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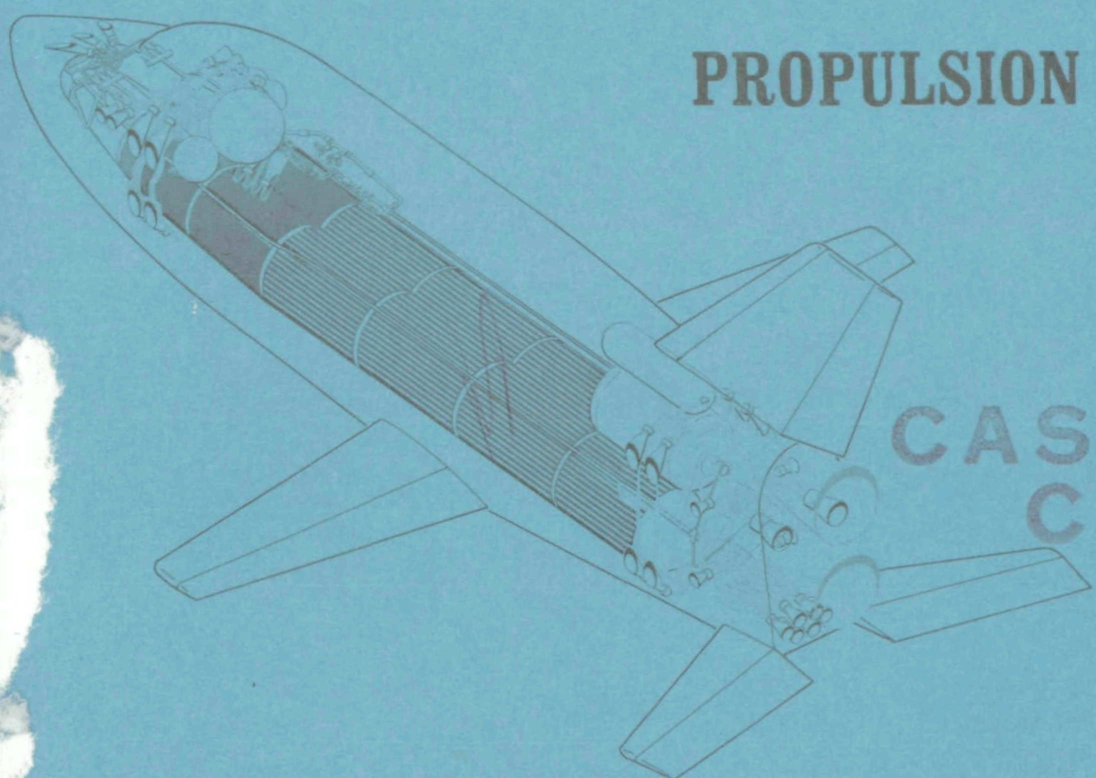
SPACE SHUTTLE LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM DEFINITION



SUBTASK A REPORT

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY • EAST

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SPACE SHUTTLE LOW PRESSURE AUXILIARY PROPULSION SUBSYSTEM DESIGN DEFINITION

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SUBTASK A REPORT

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ABSTRACT

This report presents results from the first phase (Subtask A) of a study defining the most attractive low pressure, oxygen/hydrogen auxiliary propulsion subsystem (APS) for NASA space shuttle boosters and orbiters. The purpose of Subtask A was to study candidate APS concepts and select the most attractive approaches for the booster and orbiter stages of two reference space shuttles. The second and final phase of the study (Subtask B) was a preliminary design of the selected APS concepts. Results from Subtask B are presented in Report MDC E0302, and a summary of overall study effort is provided in Report MDC E0293. Finally, a design handbook containing detailed descriptions of the selected booster and orbiter APS concepts is provided in Report MDC E0301.

The Subtask A study approach through which the preferred booster and orbiter APS concepts evolved was composed of:

(1) initial screening studies to reduce the large number of design alternatives to a matrix of high value concepts, and:

(2) detailed concept comparisons and trade-offs applying a common set of criteria consistent with overall space shuttle goals and constraints.

The most important areas affecting APS concept selection were identified as the propellant storage state (liquid or supercritical), propellant thermal conditioning scheme (active heat exchanger/gas generator or passive heat exchanger), and propellant flow control (simple main engine tank mass addition or mass addition coupled with downstream pressure or pressure-temperature control). The feasibility of each approach was evaluated in terms of performance, simplicity, flexibility, development and technology to establish recommended concept selections.

The preferred APS concepts for both booster and orbiter stages of the two reference space shuttles use main engine propellant tanks as low pressure gas accumulators. The Subtask A study demonstrated that propellant residuals trapped in the main engine tanks are sufficient to meet booster APS propellant demands, and may be supplied directly from the main tanks to the engine assemblies at tank ullage pressures and temperatures. The orbiter APS requires separate liquid propellant storage tanks to supplement boost residuals. Propellants from the storage tanks are circulated through either active or passive heat exchanger assemblies (depending on the number of +x axis manuevers performed by the APS) where they are heated prior to injection into the main engine tanks. Warm propellant vapors from

main engine tanks are mixed with additional liquid propellants in downstream liquid/vapor mixers, then supplied to the engines at constant temperature and pressure (constant density). This report contains a definition of candidate APS concepts, as well as trade-offs and comparisons leading to selection of preferred concepts for the reference space shuttles.

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1. INTRODUCTION

Auxiliary propulsion will be required for space shuttle attitude and translational control. Operating with the same propellants (i.e., oxygen and hydrogen) as the shuttle main engine, these subsystems will have a minimum service life of 100 mission cycles without major overhaul or refurbishment. Two basic design approaches have been conceived for the auxiliary propulsion subsystem (APS):

(1) a high pressure concept using turbopumps or turbocompressors to achieve high operating pressure levels;

(2) a low pressure concept using main engine propellant tanks as an integral part of the subsystem, and operating at main engine tank ullage pressures.

This report deals only with the low pressure APS concept. It presents the scope and results from the first phase (Subtask A) of a two-part McDonnell Douglas Astronautics Company-East (MDAC-EAST) study effort under MSC Contract No. NAS 9-11012, titled "Space Shuttle Low Pressure Auxiliary Propulsion Subsystem Definition". The study was performed for the National Aeronautics and Space Administration, Manned Spacecraft Center (MSC), Houston, Texas, under the technical direction of Mr. N. Chaffee. A subcontract was extended to Aerojet Liquid Rocket Company to provide component weight and performance models.

The overall study objective was "to conduct preliminary auxiliary propulsion subsystem studies, which would generate information and data, for use in the space shuttle vehicle effort," and which would, "identify attractive APS concepts, define their range of applicability and limitations and identify critical technology areas and development priorities." The study was divided into two phases. The first, Subtask A, was a conceptual subsystem definition phase designed to identify APS concepts best suited to each of two reference shuttle boosters and orbiters. The second phase, Subtask B, was a preliminary design of selected subsystems to establish an in-depth understanding of subsystem design and operation. This report presents results of Subtask A studies, only. Results from Subtask B are presented in MDAC-EAST Report No· MDC E0302, and a summary of overall study effort is provided in Report No. MDC E0293. Finally a design handbook containing detailed descriptions of the selected booster and orbiter APS concepts is provided in Report No. MDC E0301.

The Subtask A study demonstrated that propellant residuals contained in the main engine tanks are sufficient to meet booster APS propellant demands and may be supplied directly from main engine tanks to engine assemblies at ullage pressures and temperatures. The orbiter APS requires separate liquid propellant storage to supplement boost residuals. Propellants from the storage tanks are circulated through heat exchanger assemblies when they are heated prior to injection into the main engine tanks. Warm propellant vapors from the main engine tanks are mixed with additional liquid propellants in a downstream liquid/vapor mixer, then supplied to the engines at constant temperature and pressure (constant density). A description of Subtask A studies leading to selection of recommended low pressure APS concepts is presented in this report. Included are the basic study approach, a summary of vehicle and subsystem requirements, identification of candidate trade study concepts, concept selections, and the study conclusions and recommendations.

2. STUDY APPROACH

The Subtask A study approach through which the preferred booster and orbiter APS concepts evolved was composed of;

(1) initial screening studies to reduce design alternatives to a matrix of high value concepts, and

(2) detailed concept comparisons and trade-offs applying a common set of criteria consistent with overall space shuttle goals and constraints.

Candidate concepts were compared in terms of performance, simplicity, flexibility, development and technology applying common guidelines, permitting identification of the most attractive concepts for further study. Studies conducted during Subtask A included the following:

(1) Establishment of subsystem schematics.

(2) Establishment of subsystem balances including subsystem, assembly and component steady-state pressures and temperatures.

(3) Preliminary definition of the environment in which the subsystem, assemblies and components must operate.

(4) Level of assembly and/or component redundancy.

(5) Identification and general description of components and assemblies.

(6) Preliminary determination of subsystem, assembly and component weights and volumes.

(7) Preliminary determination of component life requirements.

(8) Preliminary determination of engine propellant conditioning requirements.

(9) Identification of critical technology areas.

(10) Sensitivity of the conceptual subsystem to the following:

  (a) Component design O/F ratios.

  (b) Propellant storage temperature and pressure.

  (c) Fluctuations in propellant storage temperatures and pressure about the nominal values.

  (d) Engine thrust and chamber pressure.

  (e) Subsystem total impulse and single burn maximum impulse.

  (f) Duty cycle and propellant usage rate.

  (g) Minimum impulse bit.

  (h) Reentry heating.

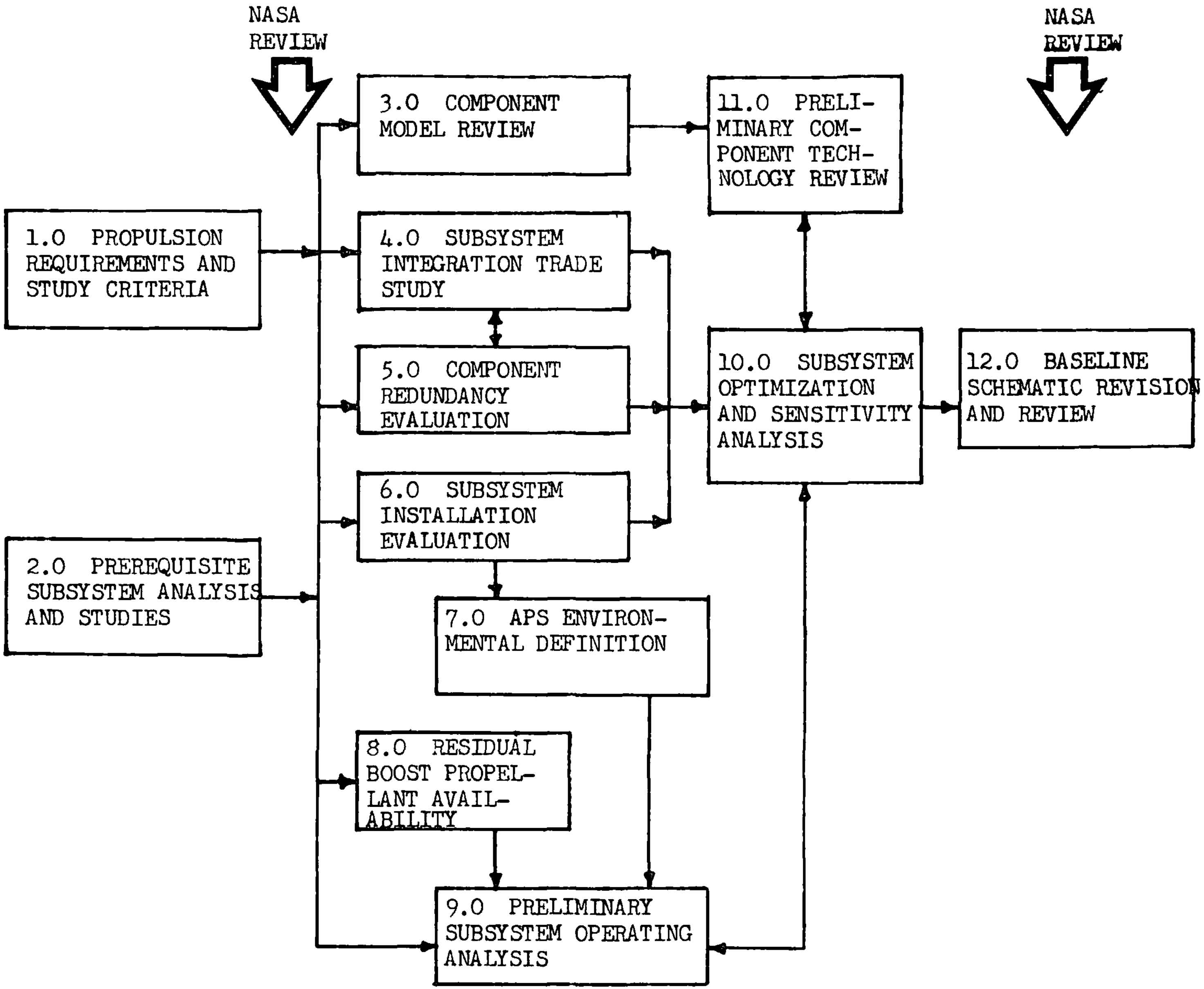
(i) Required translational and angular acceleration and velocity levels.

(j) Variations of residuals.

(k) Variations in feedline run lengths.

(l) Engines located so that no heat shield penetrations result.

A task description flow chart is provided in Figure 2-1. Initially, vehicle propulsion subsystem requirements and design criteria were established by applying the data of Appendix A. Then, for the basic low pressure APS concept, preliminary screening studies reduced the number of potential subsystem assembly candidates to a matrix of high value concepts. These assembly concepts were then evaluated and compared to identify the most attractive approaches for propellant storage, thermal conditioning, and propellant flow control. Due to the limited pressure budget for the low pressure APS concept, and the sensitivity of engine performance to injector inlet conditions, much of this effort was concentrated on propellant flow control to define operational characteristics and resulting engine performance for various control schemes. Component models were developed to assist subsystem assembly weight trade-offs and to identify key technology requirements. Finally, preferred approaches for subsystem assemblies were identified, resulting in definition of baseline subsystems for each booster and orbiter vehicle. These resulting subsystems were then weight-optimized and evaluated to determine their sensitivities to variations in vehicle design and mission operating requirements.



SUBTASK A CONCEPTUAL SUBSYSTEM DEFINITION
TASK DESCRIPTION FLOW CHART

FIGURE 2-1

3. VEHICLE AND SUBSYSTEM REQUIREMENTS

Described in this section are the primary vehicle requirements and their impact on APS design. APS engine thrust levels, number of engines, and mission total impluse allocations for each booster and orbiter are identified.

A description of baseline space shuttles considered in the Subtask A APS study and corresponding vehicle/mission requirements are provided in Appendix A. These include mission timelines, maneuver and attitude control velocity and acceleration requirements, subsystem/component design criteria, and design characteristics of the two reference space shuttles. Each of the reference shuttles consists of two reusable stages, a booster and an orbiter. Shuttle A orbiter has a low cross-range capability and is designed to reenter at a high angle of attack, to minimize vehicle heating rates and temperatures. Shuttle B orbiter has a high cross-range capability and is designed for high performance in the hypersonic flight regime. Planform drawings of both orbiters and boosters, showing APS installations selected for the Subtask A study, are presented in Figures 3-1 through 3-4. (Detailed APS installation drawings are provided in the Appendix B Configuration Analyses.)

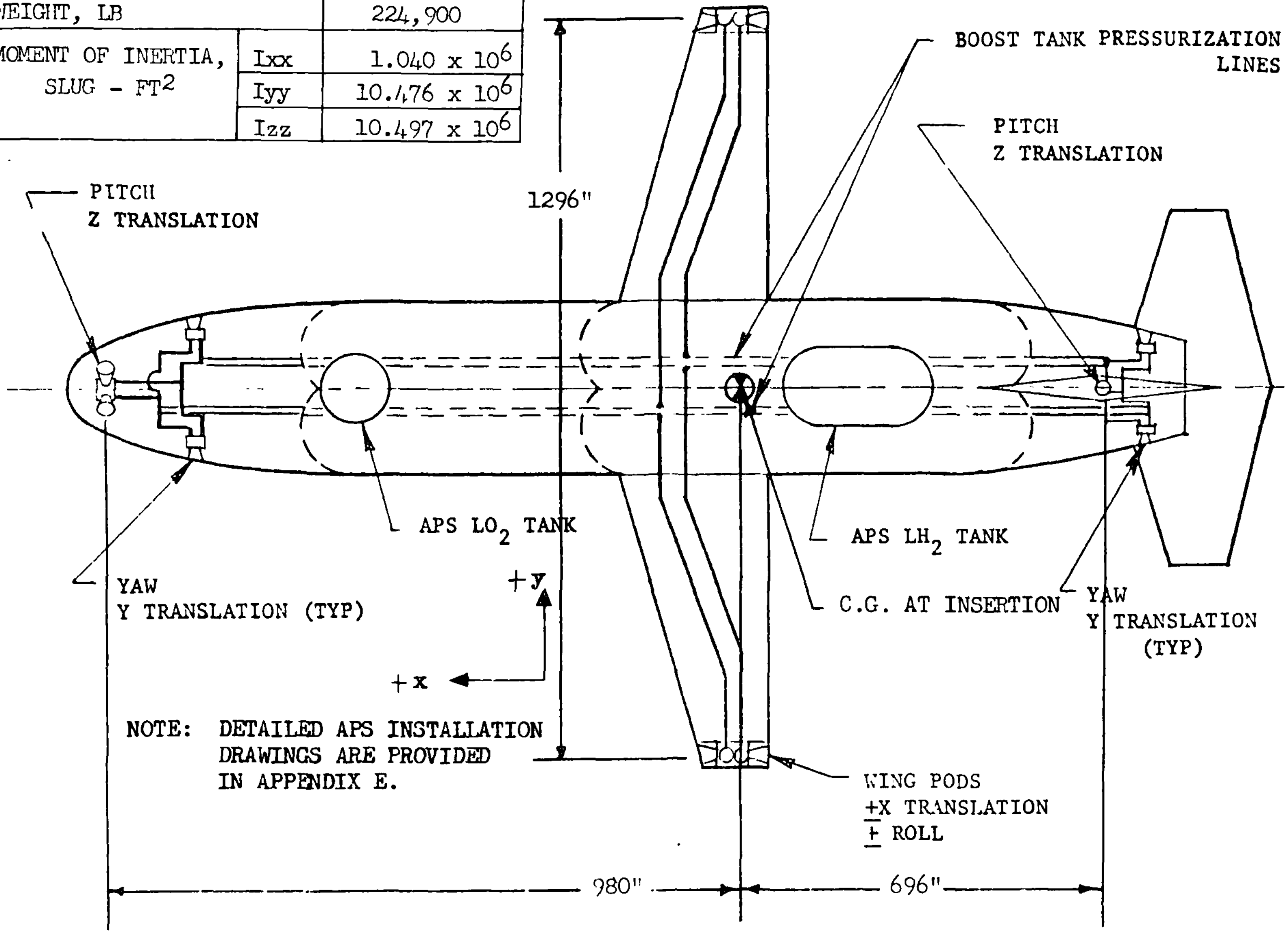
The baseline mission for both shuttles A and B was the logistics resupply of a space station or space base. Only a single mission timeline was specified for the booster, whereas three distinct mission timelines were specified for the orbiter, corresponding to the number of +X axis maneuvers to be performed by the APS. Three orbiter APS maneuver levels were considered:

(1) all translation maneuvers along all axes
(2) all translation maneuvers except those in the +X direction greater than 50 ft/sec
(3) all translation maneuvers except those in the +X direction greater than 10 ft/sec

The last two options assumed use of a separate orbit maneuver subsystem (OMS) to provide additional required maneuver ΔV.

Total impulse time histories employed in Subtask A studies are shown in Figure 3-5 for the boosters, and Figures 3-6 and 3-7 for orbiters. For those missions in which the orbiter APS provided less than the total maneuver impulse requirement, APS mission impulse included an allocation of 30,000 lb-sec per maneuver for settling liquid propellants in the OMS. The fraction of APS mission

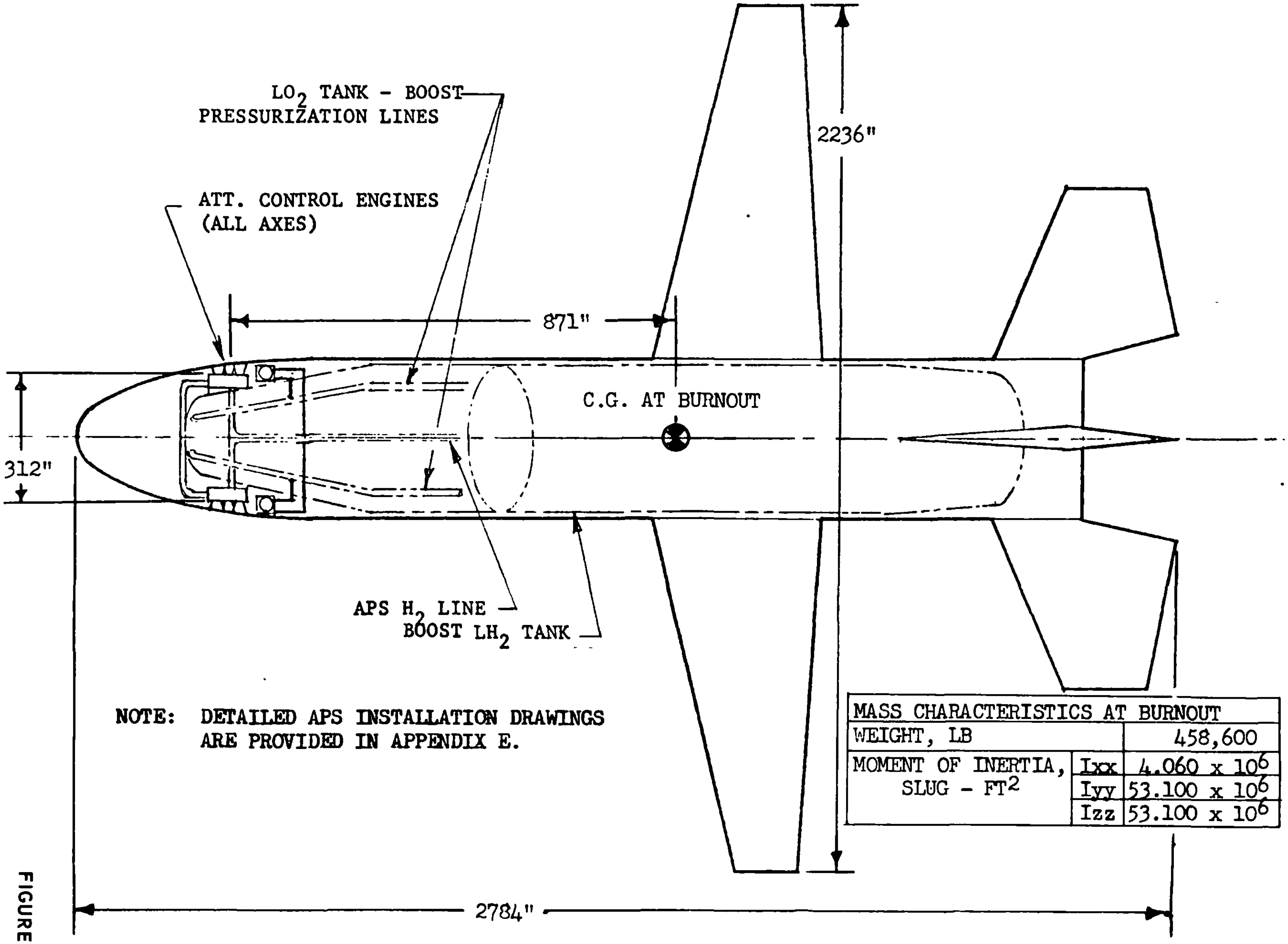
MASS CHARACTERISTICS AT INJECTION		
WEIGHT, LB		224,900
MOMENT OF INERTIA, SLUG - FT^2	Ixx	1.040×10^6
	Iyy	10.476×10^6
	Izz	10.497×10^6



VEHICLE A ORBITER

APS INSTALLATION

FIGURE 3-1



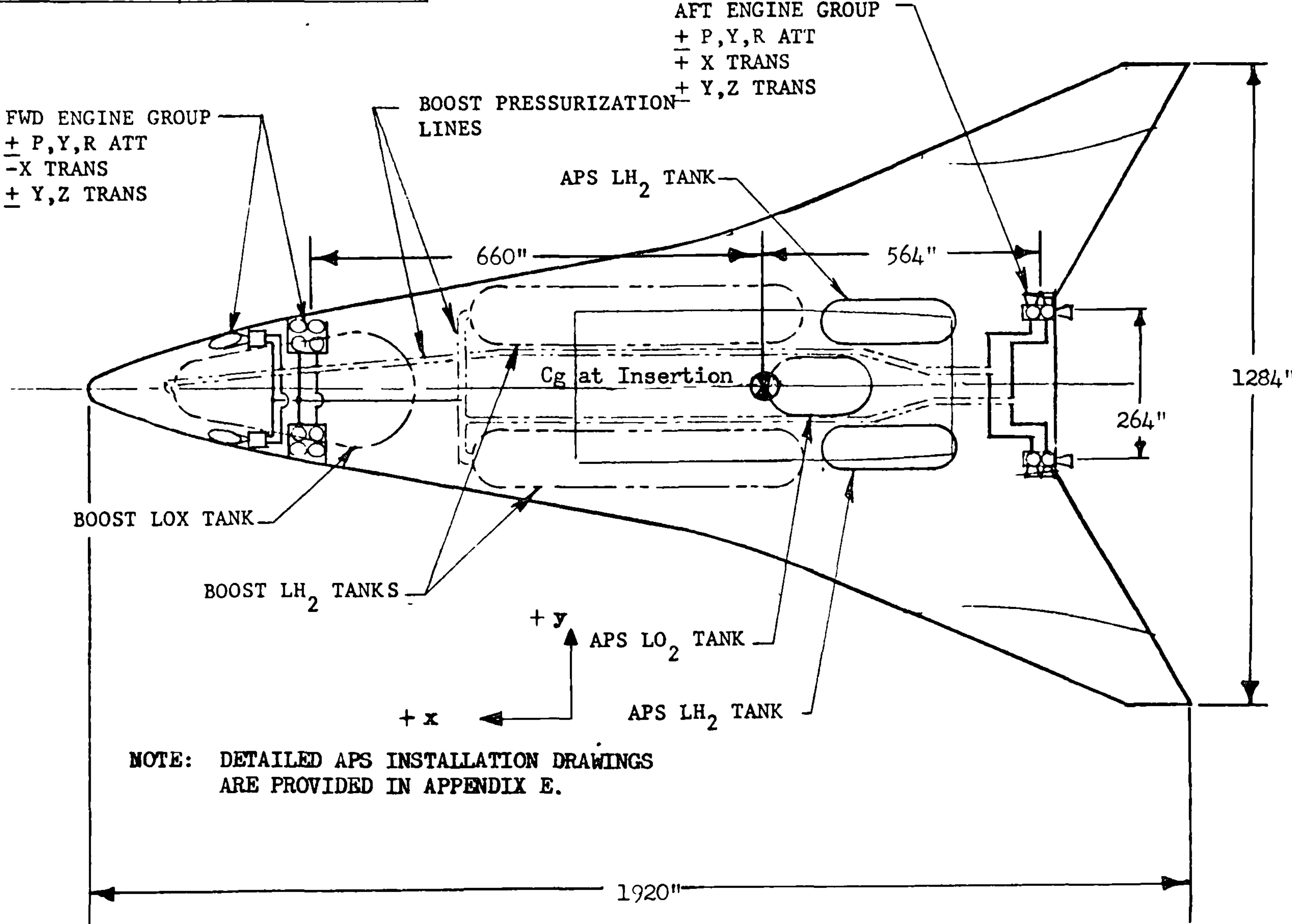
NOTE: DETAILED APS INSTALLATION DRAWINGS ARE PROVIDED IN APPENDIX E.

MASS CHARACTERISTICS AT BURNOUT		
WEIGHT, LB		458,600
MOMENT OF INERTIA, SLUG - FT^2	Ixx	4.060×10^6
	Iyy	53.100×10^6
	Izz	53.100×10^6

BOOSTER - VEHICLE A

FIGURE 3-2

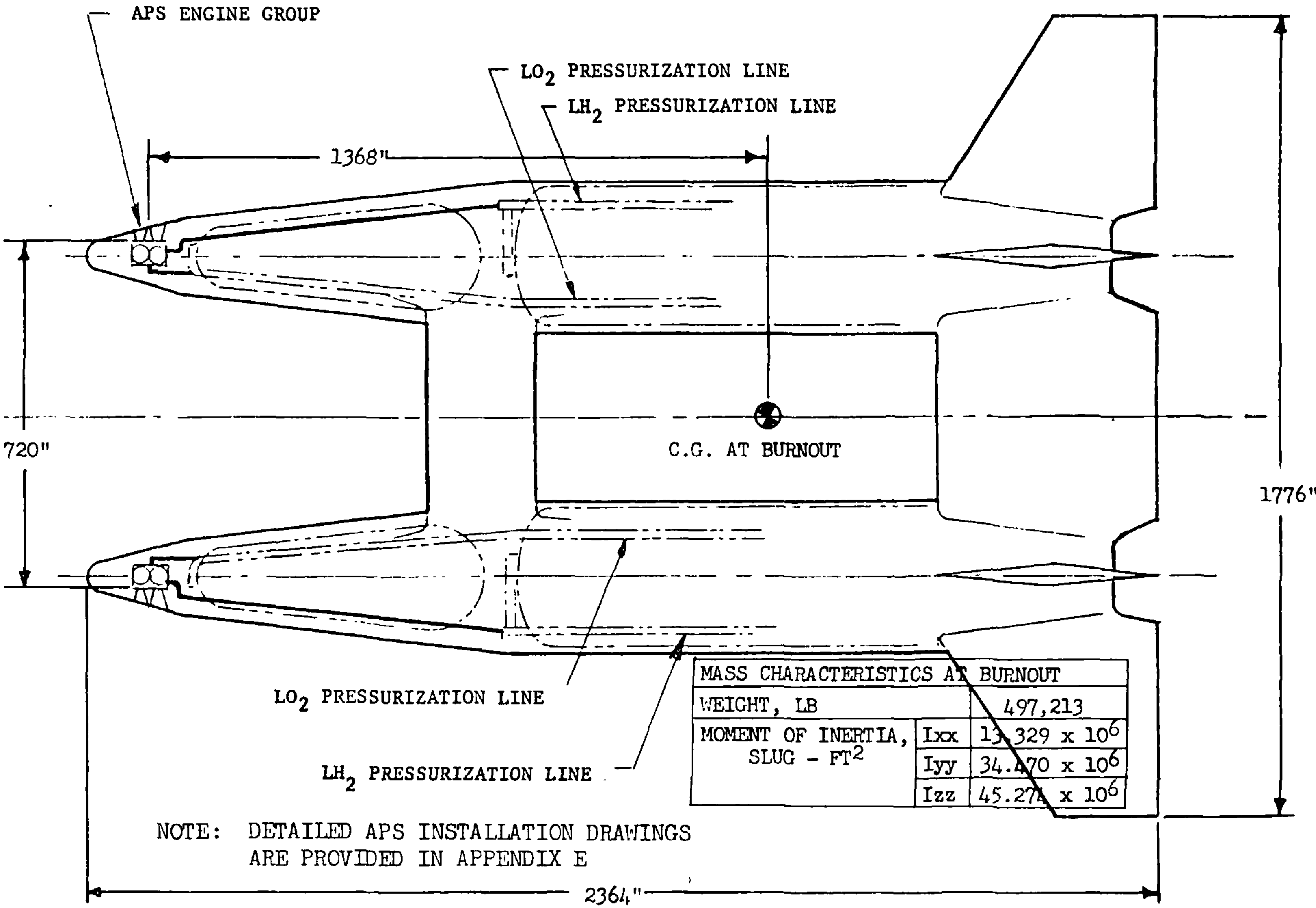
MASS CHARACTERISTICS AT INJECTION		
WEIGHT, LB		256,790
MOMENT OF INERTIA, SLUG - FT^2	Ixx	2.28×10^6
	Iyy	12.078×10^6
	Izz	13.356×10^6



NOTE: DETAILED APS INSTALLATION DRAWINGS ARE PROVIDED IN APPENDIX E.

VEHICLE B ORBITER
APS INSTALLATION

FIGURE 3-3



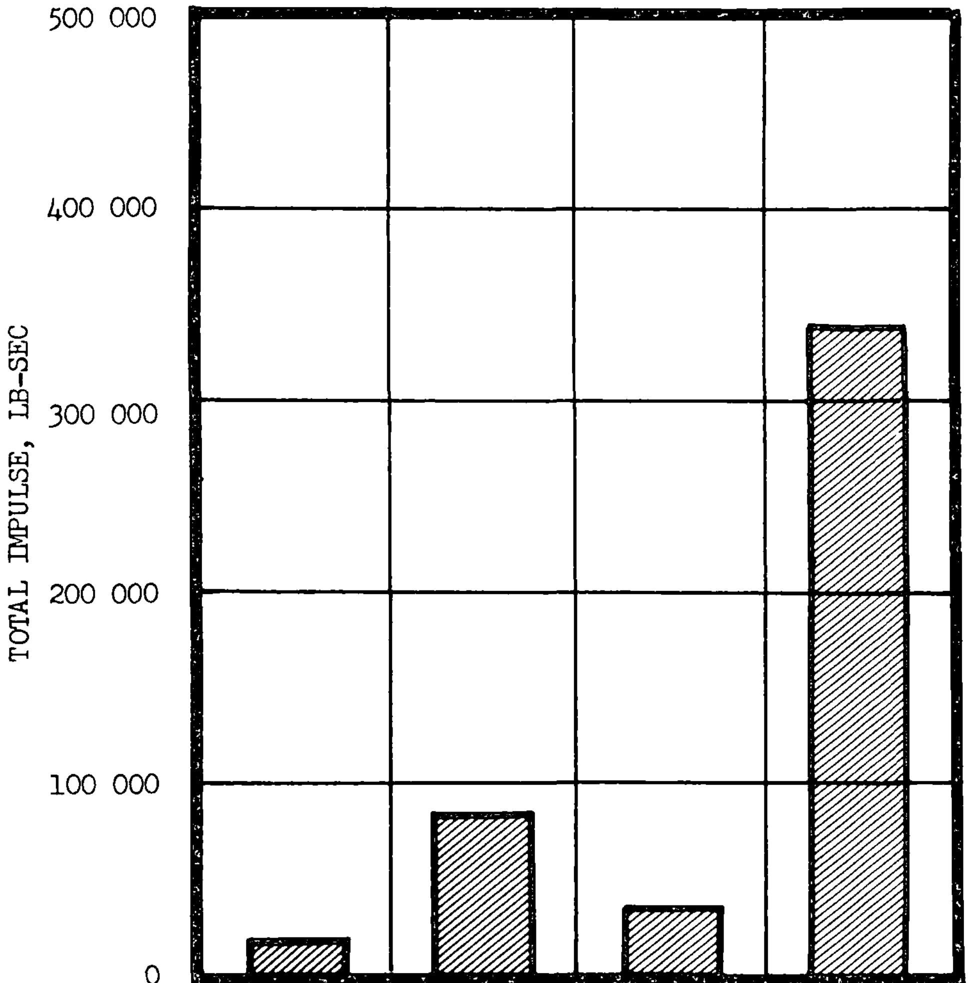
MASS CHARACTERISTICS AT BURNOUT		
WEIGHT, LB		497,213
MOMENT OF INERTIA, SLUG - FT^2	Ixx	13.329×10^6
	Iyy	34.470×10^6
	Izz	45.274×10^6

NOTE: DETAILED APS INSTALLATION DRAWINGS ARE PROVIDED IN APPENDIX E

VEHICLE B BOOSTER

APS INSTALLATION

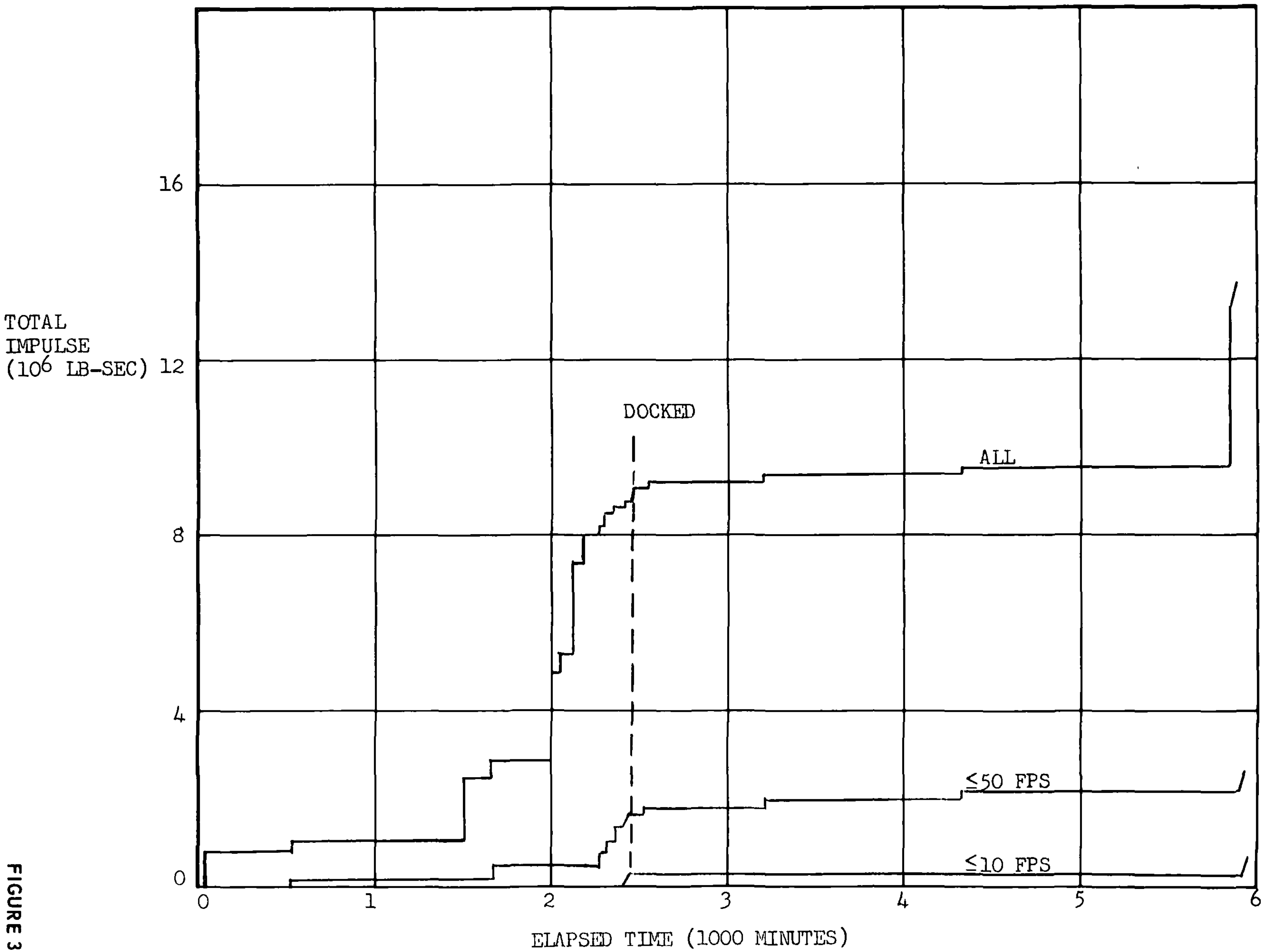
FIGURE 3-4



EVENT	POST SEPARATION	ORIENTATION	ATTITUDE HOLD	REENTRY
EVENT DURATION MINUTES		0.7 TO 0.8	0.2 TO 0.3	1 TO 5

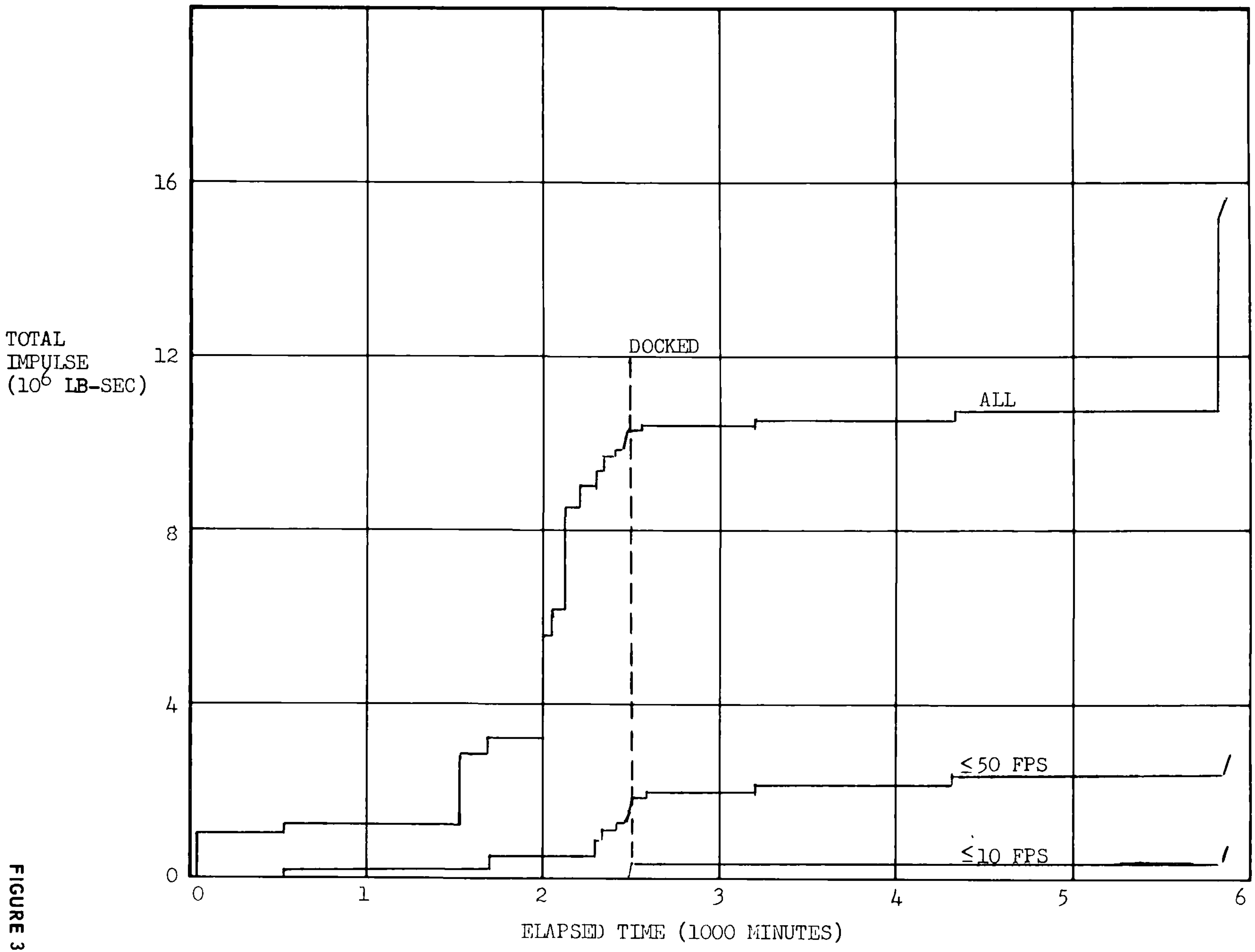
BOOSTER A AND B TOTAL IMPULSE HISTORY

FIGURE 3-5



ORBITER A IMPULSE TIME HISTORY

FIGURE 3-6

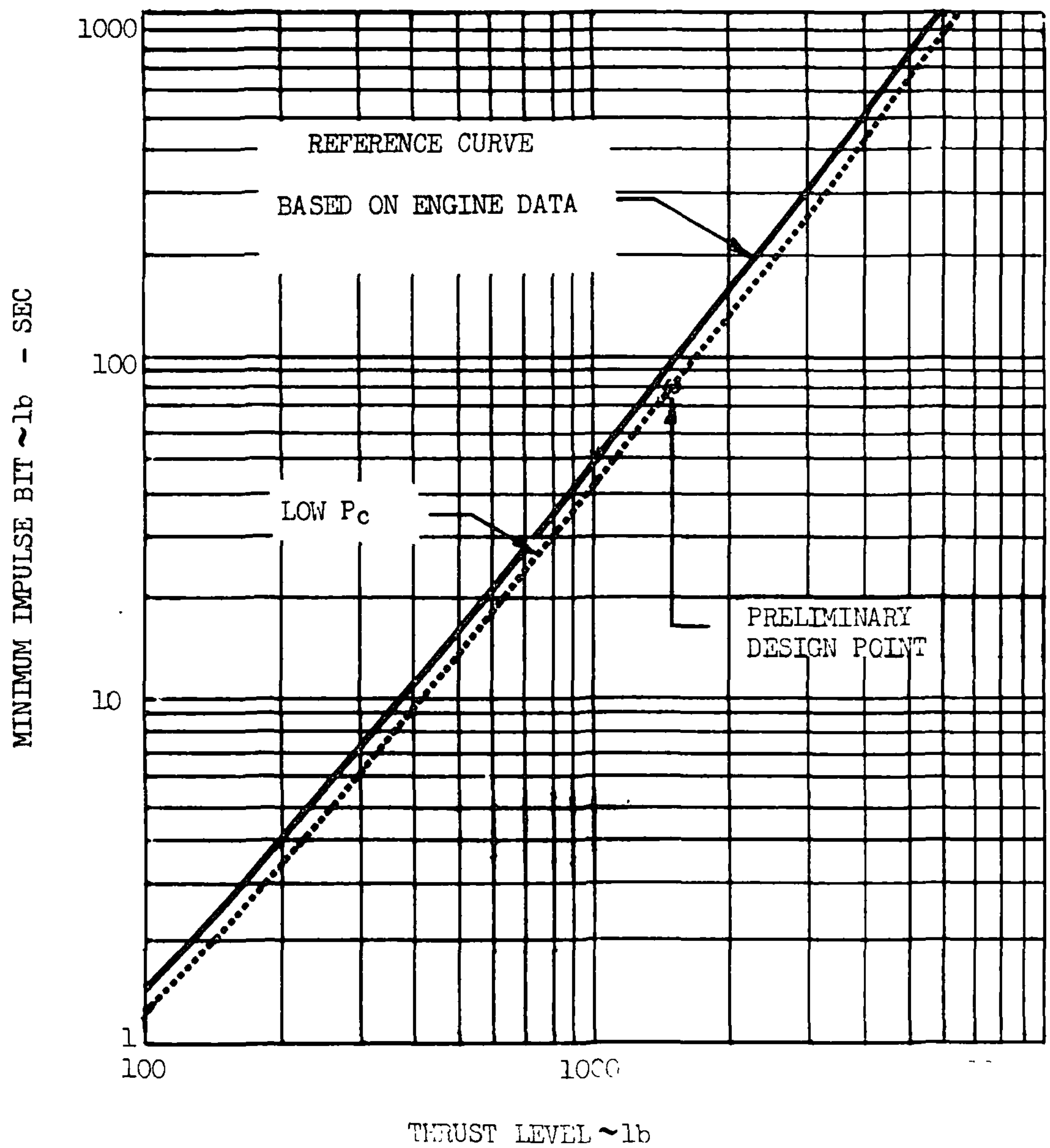


ORBITER B IMPULSE TIME HISTORY

FIGURE 3-7

impulse required for pulse-mode limit cycle operation was determined applying the minimum impulse bit-thrust correlation of Figure 3-8, in conjunction with the vehicle inertia properties and attitude deadband requirements defined in Appendix A. Figure 3-8 was derived from existing engine data for earth storable engines and calculated data for cryogenic O_2-H_2 engines.

Vehicle maneuver and attitude control requirements, specified in Appendix A, are tabulated in Figures 3-9 and 3-10 for orbiters and boosters, respectively. Two translational and angular acceleration ranges are shown. The desired range is bounded by the nominal minimum to nominal maximum values, whereas the acceptable range is bounded by the minimum and maximum values specified. These requirements were employed in conjunction with the vehicle moments of inertia, available control moment arms, and the mission failure constraints to define APS thrust levels. The ground rules used to determine total APS thrust levels are summarized in Figure 3-11. They provided for nominal minimum acceleration levels with all engines firing, and for minimum acceleration levels with two engines out. The number of engines was established so that minimum subsystem weight was achieved. The same thrust level was required for all engines to minimize development requirements. Mission performance requirements were satisfied with a booster A configuration employing eighteen 2600 lb thrust engines and a booster B configuration using sixteen 2000 lb thrust engines. Similarly, orbiter mission requirements were met with an orbiter A configuration of thirty-two 500 lb thrust engines, and an orbiter B configuration having twenty-eight 1000 lb thrust engines. Engine operational logic for these configurations are specified in Appendix B. A summary of total impulse, thrust and maximum propellant flow requirements for the booster and orbiter stages of both shuttles is presented in Figure 3-12. Propellant flow requirements were based on a maximum of five engines firing at a given time.



PRELIMINARY IMPULSE BIT DEFINITION

FIGURE 3-8

EVENT		2nd STAGE BOOST ENGINE OUT			1 TO 28 * 46 TO 53*			29 TO 45*			54*		
		X	Y	Z	X	Y	Z	X	Y	Z	X	Y	Z
TRANSLATION ACCELERATION ft/sec^2	MIN NOM MIN NOM MAX MAX		NO REQUIREMENT		0.07 0.1 0.5 1.0	0.07 0.1 0.25 1.0	0.07 0.1 0.25 1.0		SAME AS EVENT 1			NO REQUIREMENT	
		R	P	Y	R	P	Y	R	P	Y	R	P	Y
ANGULAR ACCELERATION deg/sec^2	MIN NOM MIN NOM MAX MAX				0.3 0.5 2.0 4.0	0.3 0.5 2.0 4.0	0.3 0.5 2.0 4.0		SAME AS EVENT 1		0.3 1.0 1.75 4.0	0.3 0.5 1.0 4.0	0.3 1.00 1.75 4.0
FINE ATTITUDE LIMITS - deg			NO REQUIREMENT		0.5	0.5	0.5	0.5	0.5	0.5	2.0	2.0	2.0
COARSE ATTITUDE LIMITS - deg					45	45	45	5.0	5.0	5.0			

***EVENT NUMBERS ARE DEFINED IN APPENDIX A MISSION TIMELINES.**

FIGURE 3-9

SPACE STATION/BASE LOGISTICS MISSION - ORBITER
MANEUVERING CAPABILITY REQUIREMENTS

EVENT		POST SEPARATION			ORIENTATION			ATTITUDE HOLD			REENTRY		
		X	Y	Z	X	Y	Z	X	Y	Z	X	Y	Z
TRANSLATION ACCELERATION ft/sec^2	MIN NOM MIN NOM MAX MAX			NO REQUIREMENT						NO REQUIREMENT			
		R	P	Y	R	P	Y	R	P	Y	R	P	Y
ANGULAR ACCELERATION deg/sec^2	MIN NOM MIN NOM MAX MAX	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0
ANGULAR RATE deg/sec	MIN NOM MIN NOM MAX MAX				2	2							
ATTITUDE LIMITS - deg		2	2	2	2	2	2	2	2	2	2	2	2

SPACE STATION/BASE LOGISTICS MISSION - BOOSTER
MANEUVERING CAPABILITY REQUIREMENTS

FIGURE 3-10

° ORBITER

(TOTAL THRUST LEVEL PER AXIS CALCULATED TO PROVIDE ON-ORBIT REQUIREMENTS)

	ACCELERATION					
	(FT/SEC^2)			(DEG/SEC^2)		
	X	Y	Z	PITCH	YAW	ROLL
MINIMUM NOMINAL MINIMUM	.07 .1	.07 .1	.07 .1	.3 .5	.3 .5	.3 .5

° BOOSTER

(TOTAL THRUST LEVEL PER AXIS CALCULATED TO MEET REENTRY REQUIREMENTS)

	ACCELERATION (DEG/SEC^2)		
	PITCH	YAW	ROLL
MINIMUM NOMINAL MINIMUM	.3 .5	.3 1.0	.3 1.0

NOTE: THE NUMBER OF ENGINES AND THRUST LEVEL CALCULATED TO PROVIDE:

(1) MINIMUM APS WEIGHT
(2) NOMINAL MINIMUM ACCELERATION WITH ALL ENGINES FIRING
(3) MINIMUM ACCELERATION WITH TWO ENGINES OUT

GROUND RULES FOR APS THRUST DETERMINATION

FIGURE 3-11

	ORBITER A	ORBITER B
ENGINE THRUST LEVEL, LB	500.	1000.
TOTAL NUMBER OF ENGINES	32	28
MAX. NUMBER OF ENGINES FIRING	5	5
MAX. PROPELLANT FLOW RATE, LB/SEC	$H_2 = 1.68$ $O_2 = 5.05$	$H_2 = 3.3$ $O_2 = 9.9$
TOTAL IMPULSE, 10^6 LB-SEC		
$\Delta V \leq 10$ FT/SEC	1.500	1.778
$\Delta V \leq 50$ FT/SEC	2.985	3.521
ALL MANEUVERS	13.938	16.079

	BOOSTER A	BOOSTER B
ENGINE THRUST LEVEL, LB	2600.	2000.
TOTAL NUMBER OF ENGINES	18	16
MAX. NUMBER OF ENGINES FIRING	5	5
MAX. PROPELLANT FLOW RATE, LB/SEC	$H_2 = 8.1$ $O_2 = 32.4$	$H_2 = 6.3$ $O_2 = 25.2$
TOTAL IMPULSE, 10^6 LB-SEC	.475	.475

APS REQUIREMENTS SUMMARY

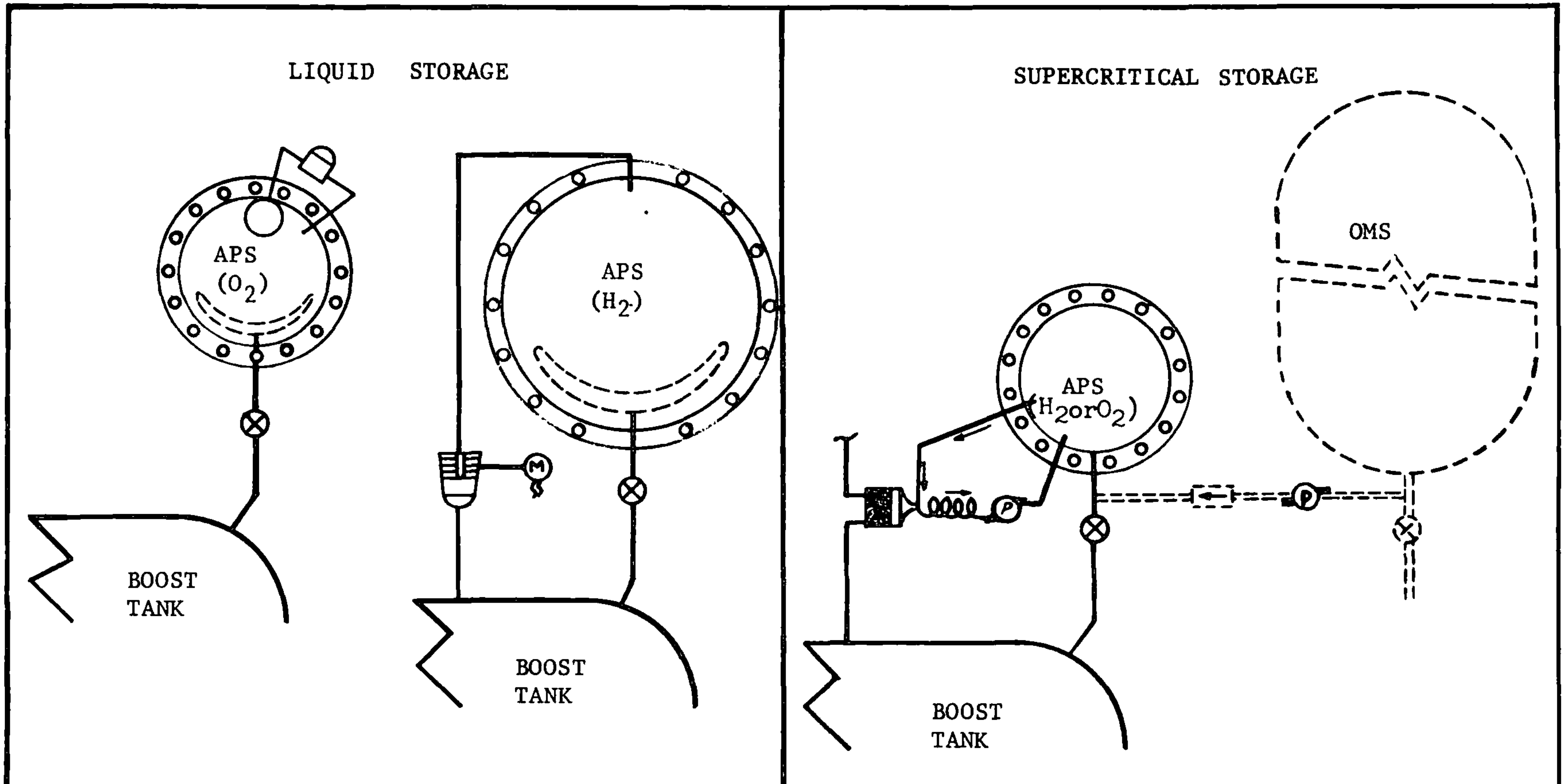
FIGURE 3-12

4. TRADE STUDY CONCEPTS

At the start of the Subtask A study, numerous potential candidates for the various APS assemblies and components were conceived. The initial problem was to reduce the number of candidates to a matrix of high-value concepts so that the study could proceed on a timely basis. Therefore, preliminary screening studies (Appendix C) were conducted on both an assembly and a component level to identify the most attractive concepts. The major concept drivers were identified as the propellant storage state (liquid or supercritical), propellant conditioning scheme (active or passive), and propellant flow control (simple main engine tank mass addition or mass addition coupled with downstream pressure or pressure-temperature control). The following paragraphs discuss each design approach. Detailed analyses, conducted to establish the most competitive position for each concept, are presented in Appendices C through F of this report. In all, sixty-eight concepts were chosen for continued examination. Selected component types and main engine tank residual propellant utilization study results are also summarized in this section, since they impact subsequent concept comparisons.

4.1 Propellant Storage - Competing propellant storage concepts which were established for each of three orbiter mission timelines (+X axis maneuvers $\leq$ 10 ft/sec; +X axis maneuvers $\leq$ 50 ft/sec; and all +X axis maneuvers) are illustrated in Figure 4-1. As this illustration shows, both liquid and supercritical propellant storage options were considered competitive for the low and intermediate velocity level missions. However, because of the extreme weight penalty (>10,000 lb) associated with supercritical storage, only liquid storage was considered a viable candidate for the high maneuver level mission. Different pressurization concepts were employed for the liquid oxygen and hydrogen storage tanks. The liquid oxygen tank employed a conventional cold-gas helium pressurization assembly; the liquid hydrogen tank was pressurized by warm hydrogen gas extracted from the main engine tank and compressed to the required ullage pressure. This latter concept was selected as a result of an overriding weight advantage. In both cases, the liquid oxygen and hydrogen tanks were nonrefillable and separate from the OMS. The overall weight advantage of separate tanks was slight, however, compared to the fully integrated APS/OMS tankage approach.

Supercritical propellant storage was an attractive concept because the propellant is stored in a single phase, eliminating the need for a propellant acqui-



- CANDIDATE FOR ALL 3 LEVELS OF + X AXIS MANEUVERS

- NON-REFILLABLE FOR + X $\Delta V \leq 10$ fps
- REFILLABLE FROM THE OMS FOR + X $\Delta V \leq 50$ fps

SELECTED TANKAGE CONCEPTS

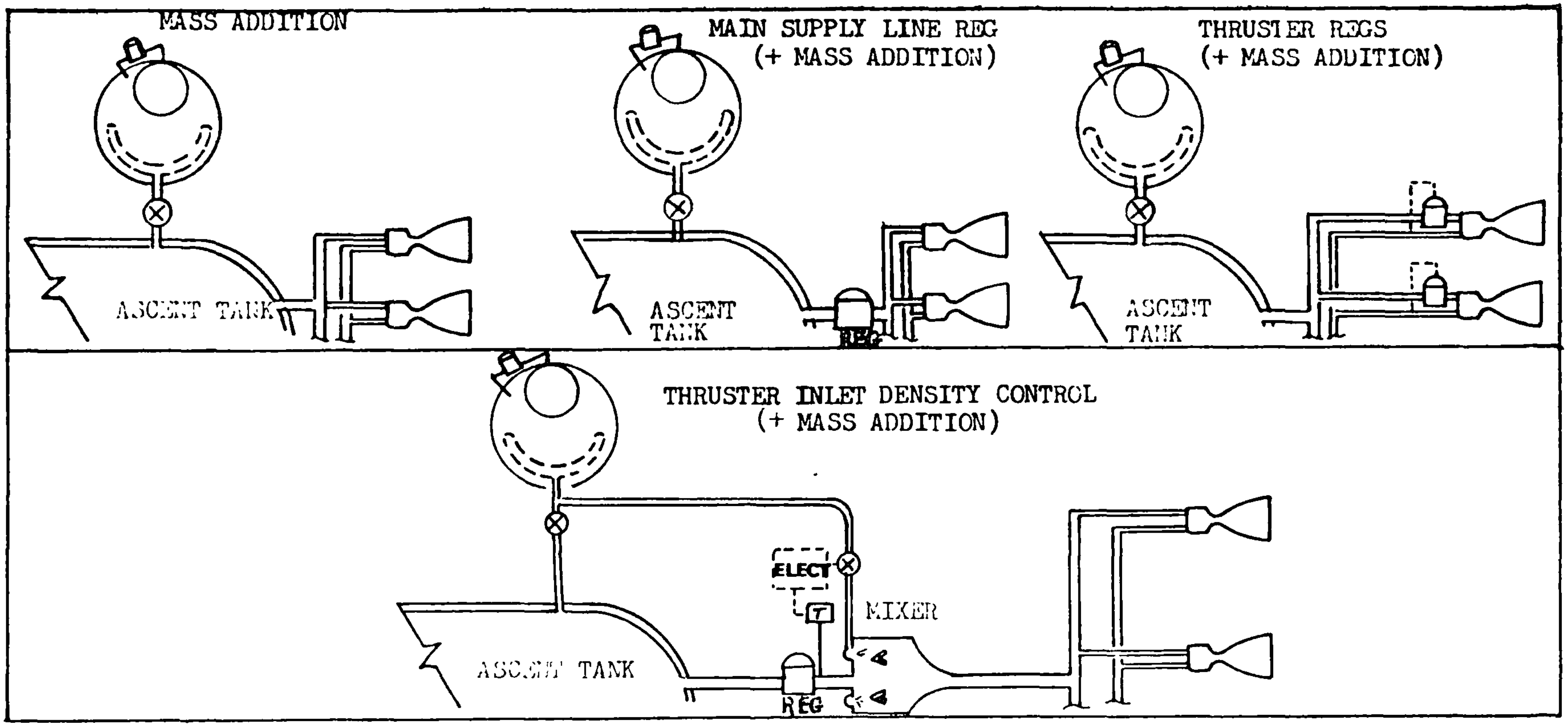
FIGURE 4-1

sition device. However, the required storage pressures are high (250 to 800 lbf/in^2a for hydrogen and oxygen, respectively) resulting in a significant weight penalty for large tanks. The supercritical storage concept shown in Figure 4-1 was separate from the OMS. For the low maneuver level mission, nonrefillable tanks were employed; however, for the intermediate level mission, refillable tanks were selected in order to reduce tank weight penalty. During expulsion, the supercritical propellant was heated to maintain required feed pressures. The preferred heating concept for this study employed an external active heat exchanger/gas generator circuit, wherein circulated propellant was heated to a prescribed temperature, pumped to a higher pressure, and then cycled back into the storage tank. This technique eliminated the explosion hazard that may result from the gas generator exhaust products leaking into the storage tanks and allowed the pump to operate at constant fluid inlet density.

The thermal protection concept preferred for both liquid and supercritical storage options employed reusable high performance insulation (HPI) enclosed in a fiberglass outer shell. The HPI was pressurized during ascent to orbit, depressurized to vacuum on orbit, and repressurized during reentry in order to reduce the fiberglass shell weight penalty.

4.2 Propellant Flow Control - Propellant flow control maintains engine thrust and mixture ratio within prescribed limits. Four different approaches were selected for further study for the orbiters, whereas only two were selected for the boosters.

4.2.1 Orbiter Propellant Flow Control - The four flow control concepts considered for the orbiters are illustrated in Figure 4-2. The concepts provided progressively greater accuracy in control of engine inlet conditions. The first concept evaluated was propellant mass addition. Main engine tank pressure was allowed to decay during APS usage until a prescribed level was attained. The main engine tanks were then resupplied with propellant from the APS storage tanks to maintain minimum engine feed pressures. This approach provided limited engine thrust and mixture ratio control. Only indirect control of engine inlet pressure and temperature was exercised through resupply propellant energy level (mass and enthalpy). The second approach also employed mass addition, but used differential pressure regulators in the individual engine feedlines to control the difference between hydrogen and oxygen injector inlet pressures. In this manner, accurate control of engine mixture ratio was achieved; however, engine thrust still varied over a wide range. This wide thrust variation was controlled in the third approach by replacing the differential regulators with a main supply line pressure regulator



ORBITER PROPELLANT FLOW CONTROL CONCEPTS

FIGURE 4-2

near the outlet of the main engine tank. This regulator provided relatively tight control of both thrust and mixture ratio as influenced by pressure, but off-nominal excursions still occurred because of variations in gaseous propellant temperatures. Precise control of thrust and mixture ratio was achieved in the fourth approach through use of a liquid/vapor mixing chamber adjacent to the main supply line pressure regulator. In this concept liquid propellant supplied from APS storage tanks was mixed with vapor extracted from the main engine tanks. Liquid inflow to the mixer was controlled using a cavitating venturi throttle valve which maintained nearly constant mixer inlet pressure. As APS engine propellant demand and main engine tank pressure and temperature varied during the mission, liquid flow to the mixer was throttled to maintain constant downstream vapor temperature, thus ensuring constant vapor density at the engine injector inlet. This concept permitted the engines to operate at near-optimum design conditions throughout the mission, and reduced the vapor extraction rate from the main engine tanks. In this manner minimum main engine tank pressure was maintained with lower resupply propellant energies compared with the other flow control concepts.

4.2.2 Booster Propellant Flow Control - Studies presented in Appendix E demonstrated that total booster APS impulse requirements could be satisfied completely using residual propellant vapors trapped in the main engine tanks following boost, thus eliminating the need for propellant resupply (mass addition). For this reason, only three concepts were considered for propellant flow control;

(1) a simple blowndown concept, wherein no control was exercised over engine inlet conditions;
(2) a regulator concept employing differential pressure regulators in the individual engine feedlines; and
(3) a regulator concept employing main supply line pressure regulators adjacent to the main engine tanks.

This third approach was subsequently ruled out during booster APS screening studies due to excessive weight penalty associated with pressure regulators and propellant distribution lines.

4.3 Propellant Thermal Conditioning - Studies presented in Appendix C showed that thermal conditioning of orbiter main engine tank resupply propellant was necessary to maintain tank temperatures and pressures above levels needed to sustain required engine thrust. Two concepts were evaluated for superheating the propellant prior to injection into the main engine tank;

(1) an active approach using a heat exchanger/gas generator assembly; and

(2) a passive approach employing a heat-sink type heat exchanger.

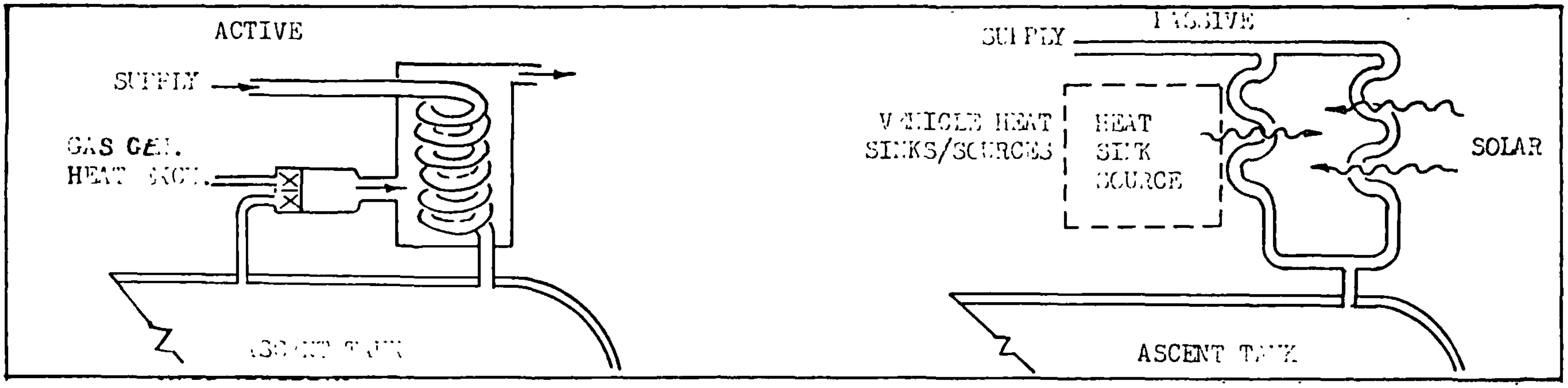
Both are illustrated in Figure 4-3. Energy for the active concept was supplied by a bipropellant gas generator, whereas energy required by the passive concept was provided by the internal energy of the orbiter external structure (mass and internal energy) and the incidental solar radiant energy flux. These two concepts were evaluated in Appendix C for each of the three orbiter mission timelines, and the following conclusions were reached:

(1) For the all-maneuver mission (i.e., when the APS performs all post-separation maneuvers), passive heat exchanger weight and vehicle surface area requirements were prohibitive. Only the active heat exchanger concept, therefore, was considered practical for this mission.

(2) For low and intermediate maneuver missions (i.e., those for which the APS performs all +X axis maneuvers of 10 ft/sec or less, and all +X axis maneuvers of 50 ft/sec or less) both active and passive conditioning concepts were weight competitive. Thus both were considered attractive candidates.

4.4 Engine Assemblies and Subsystem Components - Preliminary design and operational analyses of engine assemblies and major subsystem components were alsc conducted to define component baselines for Subtask A subsystem trade studies. Results of these analyses are presented in Appendix D, and include a description of the selected components, parametric weight and performance data used for APS modeling. Concept selections for the primary subsystem components are summarized below.

The selected engine concept consists of a coaxial element injector, fuel film cooled thrust chamber, electrical spark igniter, and diaphragm type propellant control valves. The gas generator assembly concept employs a cylindrical combustion chamber, a coaxial element injector similar to that for the engine, and electro-pneumatically actuated poppet propellant control valves. The basic active heat exchanger concept selected for both the H_2 and O_2 propellant networks was a concentric helical tube and shell design, using hot combustion gases from the gas generator assembly to heat propellant flowing in the tubes. The approach



ORBITER PROPELLANT THERMAL CONDITIONING CONCEPTS

FIGURE 4-3

selected for gaseous propellant pressure regulation was an iris-type pressure regulator, employing a series of overlapping segments moved to control propellant flow area (a concept similar to the aperture of a camera).

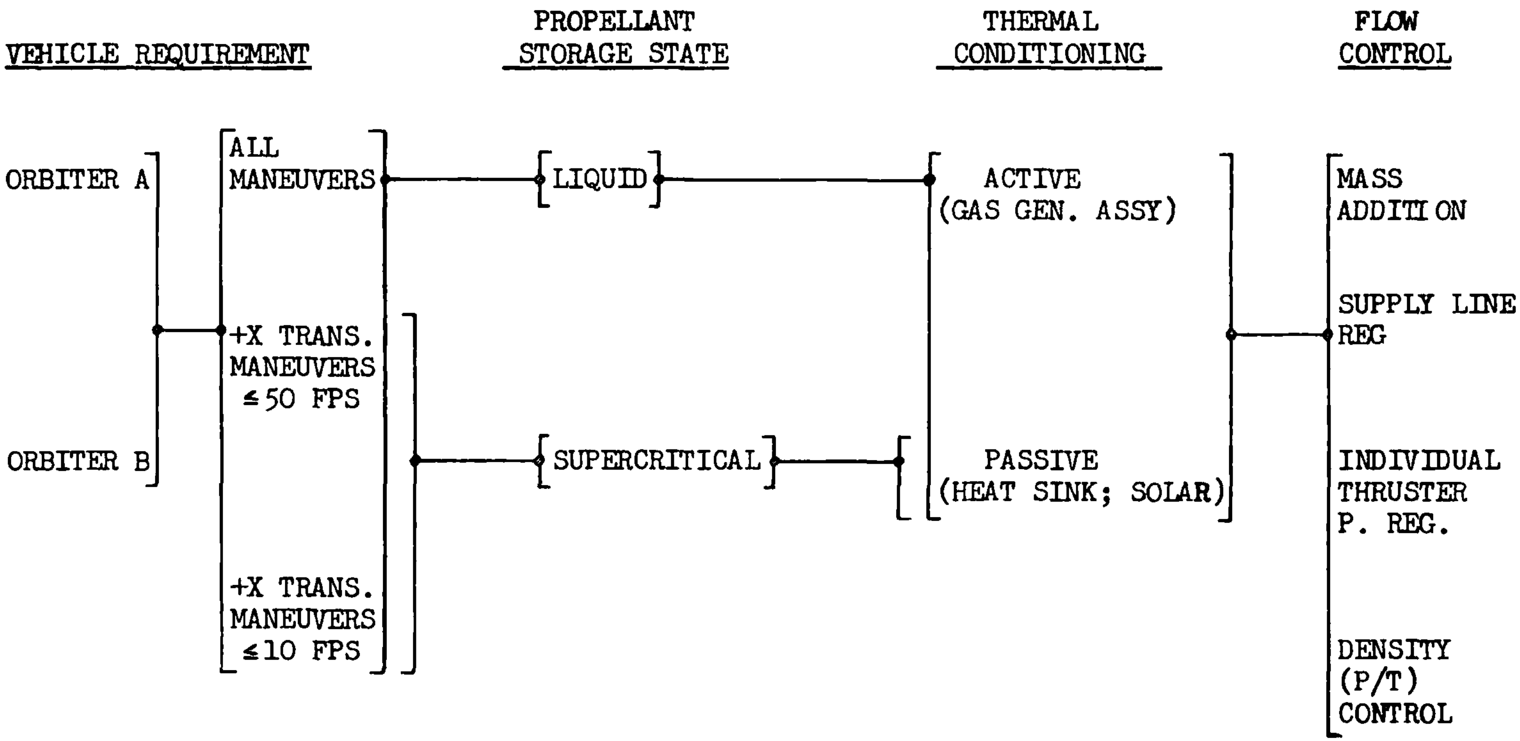
4.5 Residual Propellant Utilization - An important feature of the low pressure APS concept is its potential use of main engine tank residual propellant vapors and liquids trapped in tanks and propellant feedlines following boost. An evaluation of the usability of propellant vapors in the tank is relatively straightforward; it is based on assumed thermal environments and inventory of tank pressures and temperatures during the mission, applying the classical laws for conservation of mass and energy. A determination of residual liquid propellant usability is far more complex, however, since it depends upon propellant capillary effects, vehicle acceleration loads, main engine tank thermal environments, and APS propellant usage rates. For the purpose of obtaining a reasonably simple, but realistic model for predicting residual liquid vaporization rates, orbital flight data from Saturn IV and IVB flights were correlated to define a vaporization model which could be applied to the space shuttle. A discussion of this data correlation and application to the shuttle is presented in Appendix E. Results for the orbiter are summarized in Figure 4-4. As shown, only a small percentage of available LOX residual is usable by the APS because APS usage and oxygen boiloff were not coincident. Most oxygen boiloff was vented before major APS usage was required. Studies for the booster showed that mission total impulse requirements could be satisfied entirely using residual propellant vapors, and thus, no propellant resupply was required.

4.6 Candidate Concept Matrix - The studies discussed in the above paragraphs reduced the number of potential APS candidates to a manageable number of high value concepts having competitive characteristics. These are summarized by the matricies of orbiter and booster concepts shown in Figures 4-5 and 4-6. (More in-depth concept matricies showing design alternatives considered during APS concept screening studies are presented in Appendix F).

VEHICLE	PROPELLANT	LIQUID RESIDUAL (LB)	APS USABLE		
			≤ 10 FPS	≤ 50 FPS	ALL MANEUVERS
ORBITER A	O_2	1,827	262	365	643
	H_2	510	125	163	362
ORBITER B	O_2	670	75	118	139
	H_2	31	0	27	31

APS UTILIZATION OF MAIN ENGINE TANK LIQUID RESIDUALS

FIGURE 4-4



ORBITER CONCEPT MATRIX

FIGURE 4-5

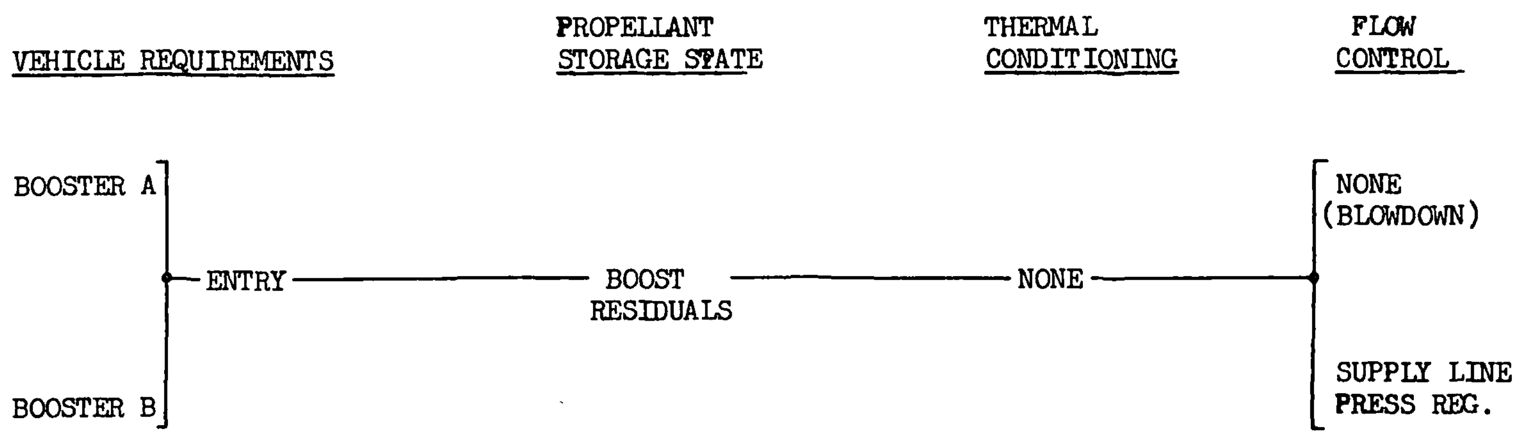


FIGURE 4-6
BOOSTER CONCEPT MATRIX

5. CONCEPT SELECTION

APS assembly concepts defined in Figures 4-5 and 4-6 for orbiters and boosters were compared in order to establish recommended baseline subsystems for subsequent Subtask B evaluations (Reference (b)). APS assemblies considered were propellant storage, thermal conditioning, and propellant flow control. To ensure fair and valid comparison, the candidate concepts were gauged to a common set of criteria consistent with space shuttle requirements and goals. These selection criteria were:

1) subsystem weight and volume
2) flexibility to mission changes
3) technology requirements
4) subsystem simplicity, and
5) development program requirements

Weighting applied to each criteria, along with the rationale used to apply the criteria for concept comparison, is presented in Figure 5-1.

To facilitate concept comparisons, the orbiter flow control assembly was considered first, holding the remainder of the system in a baseline configuration. The reference baseline was a liquid propellant storage assembly and an active heat exchanger/gas generator thermal conditioning assembly. Having selected a preferred approach for the propellant flow control function, alternate concepts for propellant storage and thermal conditioning assemblies were evaluated. Since neither a propellant storage nor a thermal conditioning assembly was required for the boosters, concept comparisons were limited to the flow control function only. Baseline APS schematics, detailed subsystem weight breakdowns, overall subsystem mission operating characteristics, and subsystem sensitivities to design and mission requirements were developed for both boosters and orbiters to conclude Subtask A definition studies. Detailed results are presented in Appendix F. The following paragraphs discuss these concept evaluation studies and the conclusions derived.

5.1 Propellant Flow Control Concept Selection - Much of the emphasis of the Subtask A definition study was placed on evaluation of orbiter propellant flow control concepts, because the flow control technique had a strong impact on engine performance, propellant thermal conditioning requirements, and total subsystem capability.

SELECTION CRITERIA	WEIGHTING (% OF TOTAL)	RATIONALE TO BE USED FOR WEIGHTING
SUBSYSTEM WEIGHT AND VOLUME	0 - 25%	WEIGHTING BASED ON ABSOLUTE WEIGHT AND VOLUME CONSIDERING LOWEST WEIGHT SYSTEM AS REFERENCE AND A 10% ORBITER PAYLOAD LOSS AS UNACCEPTABLE.
FLEXIBILITY TO MISSION CHANGES	0 - 25%	WEIGHTING BASED ON SENSITIVITY OF SUBSYSTEM TO CHANGES IN: (1) MISSION IMPULSE USAGE RATES AND TOTAL IMPULSE, (2) TEMPERATURE ENVIRONMENT, (3) CONTROL ACCELERATION REQUIREMENTS AND, (4) COMPONENT LOCATION CHANGES.
TECHNOLOGY REQUIRED	0 - 20%	WEIGHTING BASED ON ENGINEERING JUDGEMENT OF DEVELOPMENT RISK SCALING FROM STATE-OF-THE-ART THROUGH EXTENSION OF AN EXISTING TECHNOLOGY BASE TO COMPLETELY NEW CONCEPTS OR APPROACHES.
SUBSYSTEM SIMPLICITY	0 - 15%	WEIGHTING BASED ON CONSIDERATION OF: (1) THE NUMBER OF COMPONENTS AND INTEGRATION COMPLEXITY, (2) ASSEMBLY AND SUBSYSTEM CONTROL REQUIREMENTS, AND (3) COMPLEXITY OF SUBSYSTEM INTERFACES AND OPERATION.
DEVELOPMENT PROGRAM	0 - 15%	WEIGHTING BASED ON ESTIMATES OF DEVELOPMENT TEST REQUIREMENTS CONSIDERING FACTORS SUCH AS NEED FOR ENVIRONMENT SIMULATION (ZERO g, VACUUM, ETC.) AND FACILITY AVAILABILITY FOR TEST.

FIGURE 5-1

APS STUDY CONCEPT SELECTION
RECOMMENDED SELECTION CRITERIA AND WEIGHTING FACTORS

Orbiter - Prerequisite to comparing orbiter subsystem concepts, APS mission operation was investigated to identify optimum main engine tank resupply flow rates, conditioning temperatures, and minimum main engine tank pressures. This was necessary in order to establish the quantity of propellant which must be supplied or tanked in addition to the residuals from the main engine tanks to satisfy APS mission total impulse demands. These studies were conducted for each candidate orbiter propellant flow control concept.

To clarify the approach employed in performing these studies, an example is provided in Figures 5-2 through 5-5 for shuttle A orbiter APS using main supply line pressure regulators for propellant flow control. Initially, simulated mission duty cycles were analyzed with arbitrarily selected resupply flow rates and conditioning temperatures. This allowed identification of the critical APS mission phase where minimum main engine tank pressure was attained.

An example mission duty cycle is provided in Figure 5-2(a), where oxygen tank pressure is plotted as a function of time for the intermediate velocity level mission (+x axis maneuvers $\leq$50 ft/sec). As shown in Figure 5-2(a), the most critical event occurred approximately 28 hours into mission, where main engine tank pressure decayed below 24 lb/in^2a. Using the critical event identified in this manner, parametric investigation of propellant resupply flow rate and temperature was conducted holding constant the conditions unique to the critical event (i.e. total impulse, initial main engine tank pressures and temperatures). Figures 5-2(b) and 5-2(c) illustrate how variations in resupply propellant conditioning temperature and flowrate effect main engine tank pressure during the critical event. For an O_2 resupply flowrate equal to the total engine flowrate, increases in resupply temperature reduced the pressure decay, and hence extended burn time capability. A similar result was achieved if resupply temperature was held constant and ratio of resupply to engine flowrate increased. The problem, therefore, was to identify optimum ratio of resupply to engine flowrate, and then establish propellant conditioning temperature required to maintain tank pressure above a minimum prescribed level.

To find the optimum propellant resupply ratio, its effect on total propellant requirements was investigated. Using the critical event of Figure 5-2(a), Figure 5-3(a) defines the minimum conditioning temperature required to sustain a final tank pressure of 20 lb/in^2a (an arbitrary pressure level for illustration) as a function of resupply ratio. The results were substantially similar

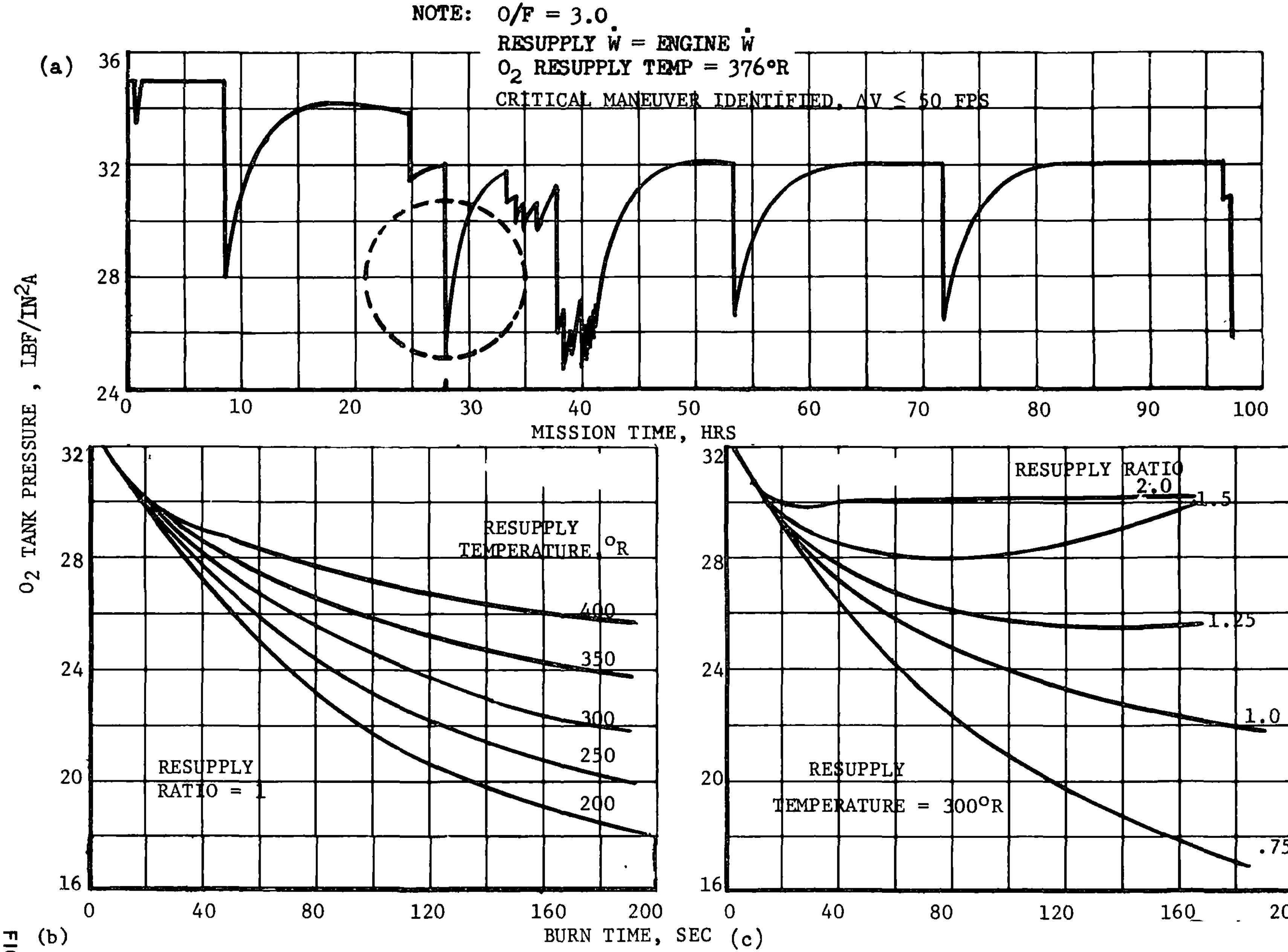


FIGURE 5-2

PROPELLANT RESUPPLY REQUIREMENTS
DEFINED BY
SINGLE BURN TOTAL IMPULSE

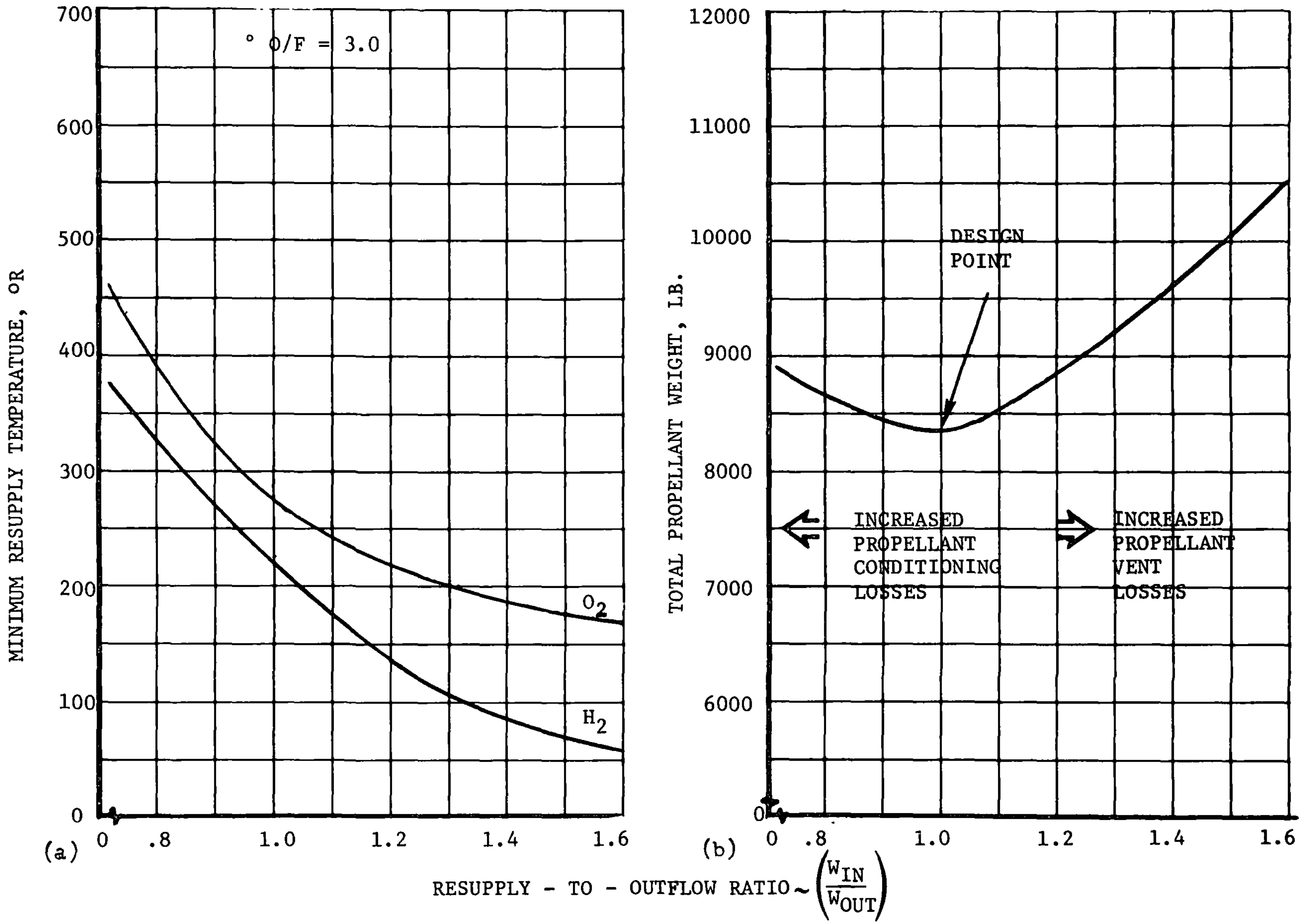


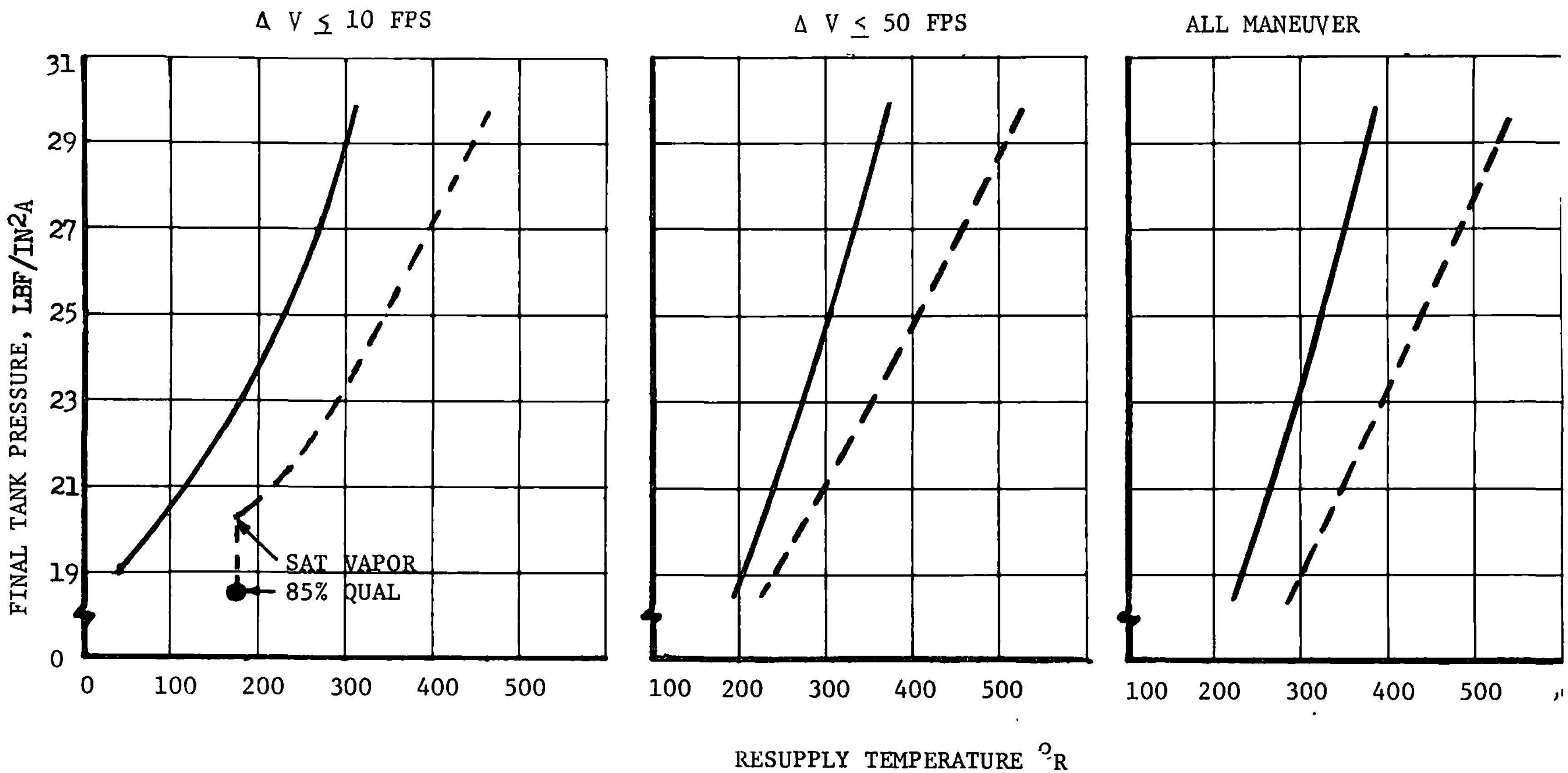
FIGURE 5-3

SELECTION OF RESUPPLY FLOW RATE

- ORBITER A, $\Delta V \leq 50$ FPS
- MAIN SUPPLY LINE PRESSURE REGULATOR
- 20 LBF/IN^{2}A MINIMUM TANK PRESSURE
- ACTIVE THERMAL CONDITIONING
- INITIAL VAPOR TEMP = 530°R
- INITIAL VAPOR PRESSURE = $\left.\begin{matrix} 43\ (H_2) \\ 33\ (O_2) \end{matrix}\right\}$ LBF/IN^{2}A

to those shown previously in Figure 5-2(c) (i.e., as resupply ratio was increased, required conditioning temperature was decreased). Figure 5-3(b) relates resupply ratio to total propellant requirement. In this example, the corresponding minimum conditioning temperatures defined in Figure 5-3(a) were employed with each resupply ratio. As shown, minimum total propellant weight was achieved for a ratio of resupply to engine flowrate of unity. This occurred because as resupply ratios were decreased below unity, conditioning temperature was increased in order to sustain tank pressure above 20 lb/in^2a. This, in turn, resulted in greater conditioning assembly weight penalties. Conversely, as resupply ratios were increased above unity, main engine tanks became overcharged and were subsequently vented to avoid overpressure. Whereas, the above results were obtained using a main supply line pressure regulator, identical results were obtained for the other flow control concepts. Therefore, for each of the four candidate flow control concepts, a resupply to engine flowrate ratio of unity was employed for concept comparisons.

With an identification of the critical event for each orbiter APS mission, and a definition of the optimum ratio of resupply to engine flowrate, conditioning temperatures required to sustain a final prescribed tank pressure during the critical event were established. Results for orbiter A are shown in Figure 5-4, where final main engine tank pressure is plotted as a function of conditioning temperature. The effect of main engine tank final pressure on overall APS weight is presented in Figure 5-5 for orbiter A. As shown, minimum subsystem weight was achieved at a final tank pressure of approximately 22.5 lb/in^2a for low and intermediate velocity level missions. However, for the high maneuver velocity mission, minimum subsystem weight would be achieved well below 20 lb/in^2a. For the purpose of providing a consistent reference, a design tank pressure of 20 lb/in^2a and the corresponding conditioning temperature were selected for the three mission velocity levels, except for those cases where higher conditioning temperatures were required to sustain main engine tank temperatures above the minimum allowable engine inlet temperatures. A tabulation of selected conditioning temperatures for each of the four candidate flow control concepts for orbiter A is provided in Figure 5-6. These conditining temperatures, and a ratio of resupply to engine flowrate of unity, were employed to define mission performance (engine thrust and mixture ratio) variations and impulsive propellant requirements. Each of the four flow control concepts were evaluated using computer simulations of the three orbiter mission duty cycles. Results of these simulations are also tabulated in



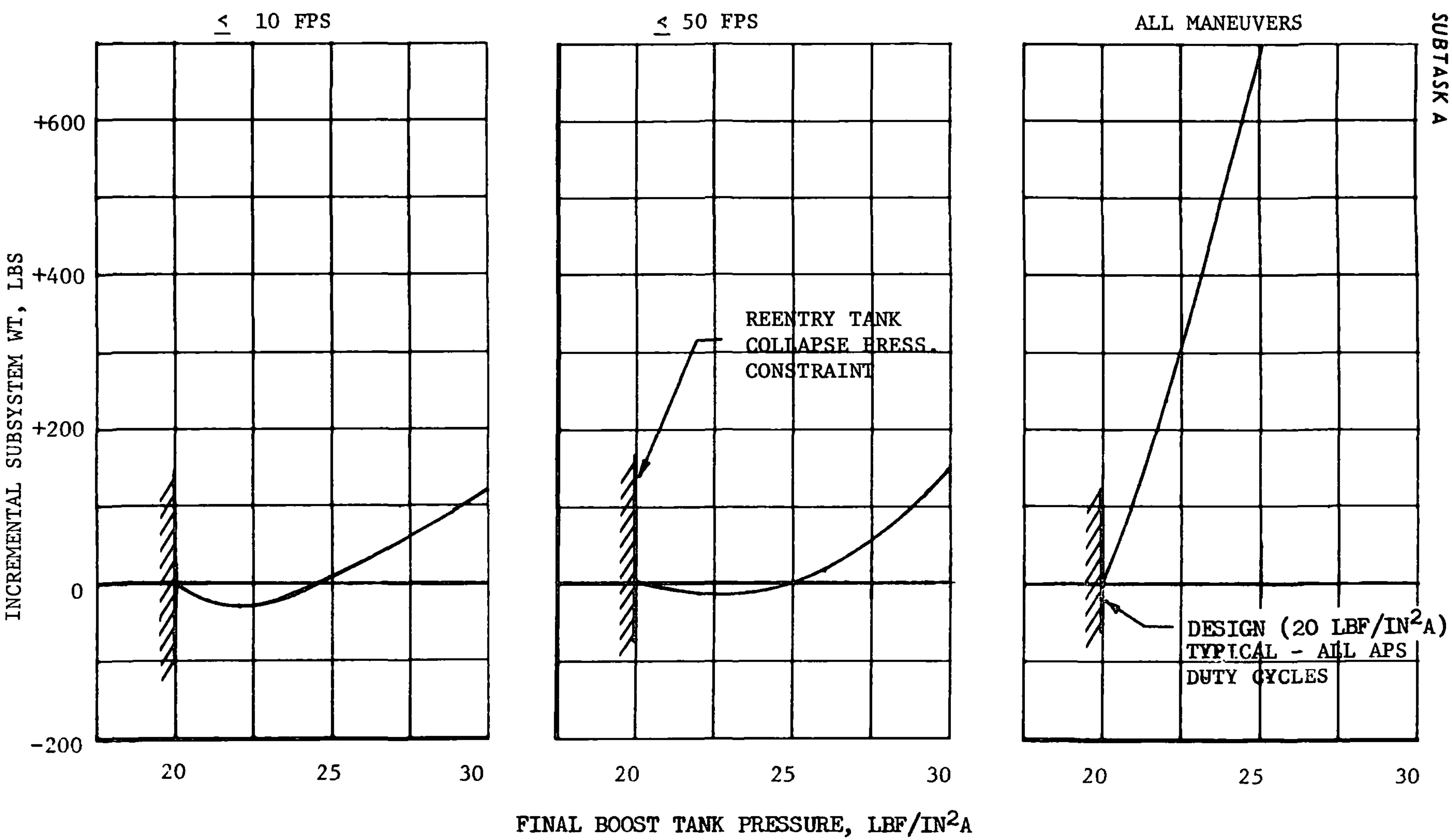
EFFECT OF RESUPPLY TEMPERATURE ON ASCENT TANK FINAL PRESSURE

- ORBITER A , O/F = 3.0
- MAIN SUPPLY LINE PRESSURE REGULATOR
- RESUPPLY $\dot{W}$ = ENGINE $\dot{W}$

  H_2

  O_2
- INITIAL VAPOR TEMP = 530°R
- INITIAL VAPOR PRESS = 43 (H_2), 33 (O_2) LBF/IN^{2}A

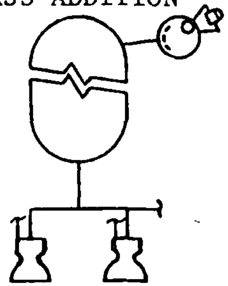
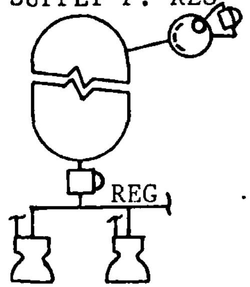
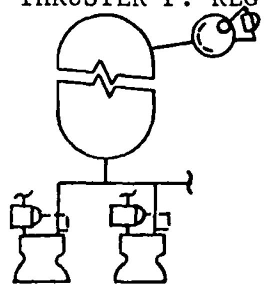
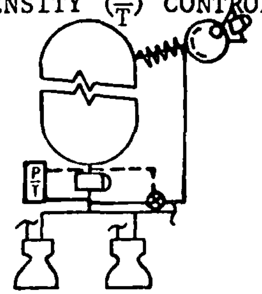
FIGURE 5-4



EFFECT OF DESIGN FINAL TANK PRESSURE ON APS WEIGHT

- ORBITER A
- LIQUID STORAGE ACTIVE CONDITIONING
- SUPPLY LINE PRESS REGULATOR
- O/F = 3.0

FIGURE 5-5

FLOW CONTROL CONCEPT	MASS ADDITION			SUPPLY P. REG			THRUSTER P. REG			DENSITY $\left(\frac{P}{T}\right)$ CONTROL		
				 REG								
MISSION	≤10	≤50	ALL	≤10	≤50	ALL	≤10	≤50	ALL	≤10	≤50	ALL
REQ'D CONDITIONING TEMP. °R (O_2)	205 *	276	326	205 *	276	326	205 *	276	326	SAT VAPOR	265	328
(H_2)	150 *	226	252	150 *	226	252	150 *	226	252	SAT LIQ	100 *	236
G.G. COND REQ'MENT (% TOTAL PROPELLANT)	9.89	14.58	16.25	9.89	14.58	16.25	9.89	14.58	16.25	3.2	5.82	10.42
ENGINE PROPELLANT REQUIREMENT (% OF TOTAL)	90.11	85.42	83.75	90.11	85.42	83.75	90.11	85.42	83.75	96.8	94.18	89.58
RELATIVE SUBSYSTEM WT	1.048	1.051	1.041	1.099	1.068	1.047	1.085	1.061	1.045	1.0	1.0	1.0

*RESUPPLY CONDITIONING TEMPERATURE SELECTED TO INSURE H_2 AND O_2 ENGINE INLET TEMPERATURES ARE MAINTAINED ABOVE 200°R, AND 300°R, RESPECTIVELY.

FLOW CONTROL PERFORMANCE COMPARISON

- o ORBITER A
- o O/F = 3.0
- o RESUPPLY $\dot{W}$ = ENGINE $\dot{W}$

FIGURE 5-6

Figure 5-6, in terms of required conditioning temperatures, gas generator propellant, engine propellant, and relative subsystem weight. As shown, propellant conditioning was required in all cases except hydrogen resupply for low maneuver level, constant density control. In that case, sufficient hydrogen conditioning was achieved by main engine tank wall-to-vapor heating.

The constant density approach affords lowest subsystem weight. This results from constant engine performance at optimum design conditions, and from reduced weight of the thermal conditioning assembly. Less gaseous propellant is extracted from the main engine tank since much of the total propellant flow is supplied as liquid to a liquid-vapor mixer for control of engine inlet temperature. This, in turn, reduces resupply flow and energy requirements for maintaining tank pressure, consequently, heat exchanger size and gas generator propellant demand are reduced. An additional advantage attributed to reduced thermal conditioning requirements is greater capability in terms of engine thrust, mission total impulse and maximum single-burn ΔV. Increases in these parameters can be accommodated for less subsystem weight penalty than for other flow control concepts. Results of the comparison for each of the four candidate orbiter flow control concepts applying previously established selection and weighting criteria, are shown in Figure 5-7. Because of minimum subsystem weight and greater mission flexibility, the constant density concept was selected as the preferred flow control approach.

Booster - Since the boosters do not require propellant resupply, only two flow control concepts were compared: a simple blowdown concept, and a pressure regulated concept using differential regulators for individual engine assemblies. Simulated mission operational characteristics of both concepts are compared in Figure 5-8 for booster A. As shown, the concepts afforded equivalent performance. Final main engine tank pressures and total engine thrust were practically identical, and engine mixture ratio remained relatively constant. Comparison of these concepts applying the selection criteria is presented in Figure 5-9. Because of minimum subsystem weight, inherent design/operational simplicity, and acceptable flexibility to mission requirements, the blowdown concept was selected for both space shuttle boosters.

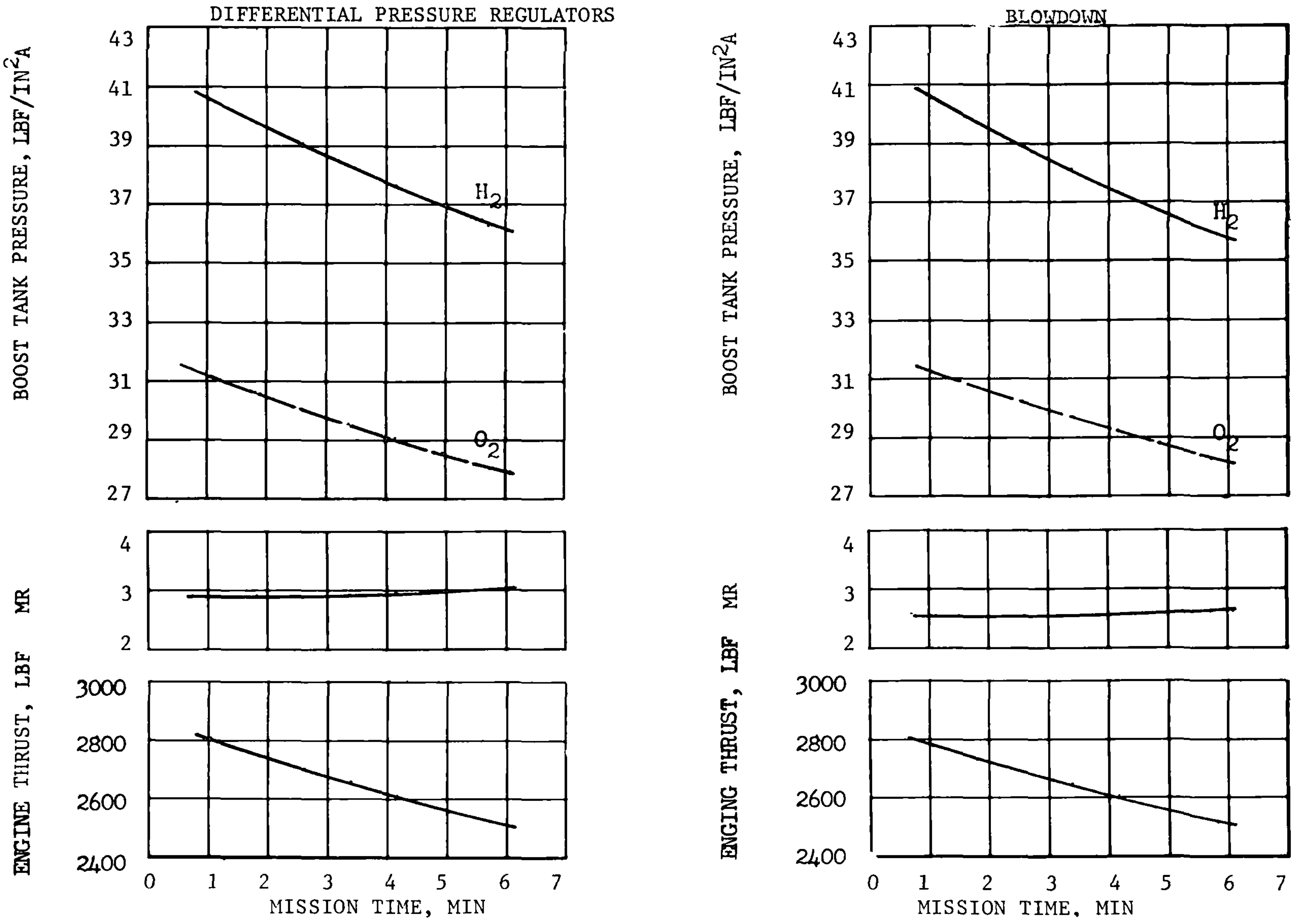
5.2 Orbiter Propellant Storage and Thermal Conditioning Concept Selection - Following selection of the constant density propellant flow control scheme, candidate propellant storage and thermal conditioning options were compared. To provide valid comparisons, each subsystem concept was weight-optimized on the

SELECTION FACTORS	MASS ADDITION	SUPPLY LINE REGULATOR	THRUSTER REGULATOR	CONST DENSITY
TECHNOLOGY (20%)				
- HARDWARE TECHNOLOGY EXTENSION (10)	10	6	8	5
- ANALYTICAL VERIFICATION (6)	6	6	6	4
- DEMONSTRATED CONCEPT (4)	4	3	3	1
	(20)	(15)	(17)	(10)
SIMPLICITY (15%)				
- NO. COMPONENTS (5)	5	3	1	2
- OPERATIONAL COMPLEXITY (5)	5	3	4	2
- SUBSYSTEM INTEGRATION (3)	3	2	2	2
- CONTROL REQUIREMENTS (2)	2	1	2	0
	(15)	(9)	(9)	(6)
WEIGHT AND VOLUME (25%)				
- WEIGHT (1 PER 100 LB)	20	17	18	25
- VOLUME (1 PER 60 FT^3)				
	(20)	(17)	(18)	(25)
MISSION FLEXIBILITY (25%)				
- SENSITIVITY TO THRUST LEVEL (10)	3	5	7	10
- SENSITIVITY TO MAX. ΔV (6)	0	2	3	6
- SENSITIVITY TO TOTAL IMPULSE (5)	3	3	3	5
- OPER. CONSTRAINTS, TIME BETWEEN BURNS (4)	0	1	2	3
	(6)	(11)	(15)	(24)
DEVELOPMENT (15%)				
- REQ'D COMPONENT DESIGN MARGINS (5)	0	2	4	5
- ENVIRONMENTAL SIMULATION (5)	2	4	5	4
- INTEGRATED TEST REQUIREMENTS (3)	1	2	3	1
- FACILITY REQUIREMENTS (2)	2	2	2	2
	(5)	(10)	(14)	(12)
TOTAL	66	62	73	77 √

√ SELECTED

FIGURE 5-7

SELECTION OF FLOW CONTROL CONCEPT



VEHICLE A BOOSTER APS OPERATIONAL CHARACTERISTICS

- INITIAL TANK VAPOR TEMPERATURES: H_2 = 200°R, O_2 = 300°R
- NO PROPELLANT RESUPPLY

FIGURE 5-8

	BOOSTER A		BOOSTER B	
	BLOWDOWN	REGULATED	BLOWDOWN	REGULATED
TECHNOLOGY (20 POINTS)				
HARDWARE TECHNOLOGY EXTENSION (10)	8	7	8	7
PROPELLANT ACQUISITION (10)	10	10	10	10
	(18)	(17)	(18)	(17)
SIMPLICITY (15 POINTS)				
NUMBER OF COMPONENTS (6)	5	2	5	2
OPERATIONAL COMPLEXITY (6)	5	3	5	3
INTEGRATION COMPLEXITY (3)	3	2	3	2
	(13)	(7)	(13)	(7)
FLEXIBILITY (25 POINTS)				
SENSITIVITY TO THRUST LEVEL (12)	12	10	12	10
SENSITIVITY TO ENTRY TIME (8)	6	8	6	8
SENSITIVITY TO BOOST HEATING (3)	3	3	3	3
SENSITIVITY TO LIQUID RESIDUALS (2)	1	2	1	2
	(20)	(23)	(22)	(23)
WEIGHT AND VOLUME (25 POINTS)				
240 LB (1PT)	25	22	25	20
	(25)	(22)	(25)	(20)
DEVELOPMENT (15 POINTS)				
ENVIRONMENTAL SIMULATION (6)	4	5	4	5
INTEGRATED TEST REQUIREMENTS (6)	6	6	6	6
FACILITY REQUIREMENTS (3)	2	2	2	2
	(12)	(13)	(12)	(13)
TOTAL POINTS	90	82	90	80

BOOSTER APS CONCEPT COMPARISON

FIGURE 5-9

basis of engine chamber pressure, mixture ratio, nozzle expansion ratio, and main engine tank pressures, then analyzed to determine sensitivity to engine thrust and mission total impulse. Results of these subsystem weight optimizations and sensitivity analyses are presented in Appendix F, along with detailed subsystem schematics. These data permitted quantitative comparison of propellant storage and thermal conditioning concepts. Summaries of detailed concept comparisons are tabulated in Figures 5-10 through 5-13 for orbiters A and B, at both low (+X axis maneuvers $\leq$10 ft/sec) and intermediate (+X avis maneuvers $\leq$ 50 ft/sec) mission velocity levels.

Based on these comparisons, a specific propellant storage and thermal conditioning concept was recommended for each orbiter and mission velocity requirement.

Referring to Figures 5-10 and 5-11 for orbiter A, and Figures 5-12 and 5-13 for orbiter B, it is seen that the liquid storage concept offered a clear advantage in all categories of comparison, with the exception of technology requirements for the low maneuver level mission, and was an obvious selection for the orbiters at both low and intermediate mission velocity levels.

The choice among candidate thermal conditioning assemblies was less obvious than that for the propellant storage state, particularly for intermediate maneuver level missions. Figures 5-10 and 5-12 (for low maneuver level missions) demonstrate the advantages of passive conditioning in the categories of technology, simplicity, and subsystem weight. In terms of subsystem technology, the passive concept possessed a clear advantage, since the heat exchanger was designed to accomodate only a single low temperature fluid. Furthermore, subsystem start-up sequencing was simpler and there were fewer assembly components. These factors were judged to override reduced environmental sensitivity and non-integrated subsystem testing afforded by the active concept. For intermediate maneuver level missions (Figures 5-11 and 5-13 for orbiters A and B, respectively) passive conditioning simplicity was compromised by additional integration complexity caused by the larger heat exchanger surface area. In addition, the design was more sensitive both to maximum velocity increment and to time required for vehicle surface temperature recovery. Based on these considerations, active and passive conditioning concepts were rated essentially even for intermediate maneuver level missions with the active conditioning concept possessing a slight point advantage due to ease of vehicle integration.

SELECTION CRITERIA	PASSIVE		ACTIVE	
	SUPERCRITICAL	LIQUID	SUPERCRITICAL	LIQUID
TECHNOLOGY (20 POINTS)				
HARDWARE TECHNOLOGY EXTENSION (10)	7	8	4	5
PROPELLANT ACQUISITION (10)	7	6	7	5
	(14)	(14)	(11)	(10)
SIMPLICITY (15 POINTS)				
NUMBER OF COMPONENTS (5)	4	5	2	3
OPERATIONAL COMPLEXITY (5)	4	5	1	2
INTEGRATION COMPLEXITY (3)	2	3	3	3
CONTROL REQUIREMENTS (2)	1	2	1	2
	(11)	(15)	(7)	(10)
FLEXIBILITY (25 POINTS)				
OPERATING CONSTRAINTS (10)	8	8	9	9
SENSITIVITY TO MAX ΔV BURN (5)	3	3	4	4
SENSITIVITY TO THRUST LEVEL (5)	2	5	2	5
SENSITIVITY TO TIME BETWEEN BURNS (3)	1	2	3	3
SENSITIVITY TO TOTAL IMPULSE (2)	1	2	1	1
	(15)	(20)	(19)	(22)
WEIGHT AND VOLUME (25 POINTS)				
80 LB (1PT)	17	25	12	24
60 CU FT (1PT)	0	0	0	0
	(17)	(25)	(12)	(24)
DEVELOPMENT (15 POINTS)				
ENVIRONMENTAL SIMULATION (6)	3	4	4	5
INTEGRATED TEST REQUIREMENTS (6)	3	3	5	5
FACILITY REQUIREMENTS (3)	2	2	1	3
	(8)	(9)	(10)	(13)
TOTAL	(65)	(83) ✓	(59)	(79)

✓ SELECTED

FIGURE 5-10

SYSTEM CONCEPT COMPARISON (P/T CONTROL)
o ORBITER A
$\Delta V \leq 10$ FT/SEC

	PASSIVE		ACTIVE	
	SUPERCRITICAL	LIQUID	SUPERCRITICAL	LIQUID
TECHNOLOGY (20 POINTS)				
HARDWARE TECHNOLOGY EXTENSION (10)	7	8	4	5
PROPELLANT ACQUISITION (10)	4	6	4	5
	(11)	(14)	(8)	(10)
SIMPLICITY (15 POINTS)				
NUMBER OF COMPONENTS (5)	4	5	2	3
OPERATIONAL COMPLEXITY (5)	4	5	1	2
INTEGRATION COMPLEXITY (3)	2	1	3	3
CONTROL REQUIREMENTS (2)	1	1	1	2
	(11)	(12)	(7)	(10)
FLEXIBILITY (25 POINTS)				
OPERATING CONSTRAINTS (10)	6	8	7	9
SENSITIVITY TO MAX ΔV BURN (5)	3	2	4	4
SENSITIVITY TO THRUST LEVEL (5)	2	5	2	5
SENSITIVITY TO TIME BETWEEN BURNS (3)	1	1	3	3
SENSITIVITY TO TOTAL IMPULSE (2)	1	2	1	1
	(13)	(18)	(17)	(22)
WEIGHT AND VOLUME (25 POINTS)				
80 LB (1PT)	23	25	18	24
60 CU FT (1PT)	0	0	0	0
	(23)	(25)	(18)	(24)
DEVELOPMENT (15 POINTS)				
ENVIRONMENTAL SIMULATION (6)	2	4	3	5
INTEGRATED TEST REQUIREMENTS (6)	2	3	4	5
FACILITY REQUIREMENTS (3)	2	2	1	3
	(6)	(9)	(8)	(13)
TOTAL POINTS	64	78	58	79 ✓

✓SELECTED

FIGURE 5-11

SYSTEM CONCEPT COMPARISON (P/T CONTROL)
ORBITER A
$\Delta V \leq 50$ FT/SEC

SELECTION CRITERIA	PASSIVE		ACTIVE	
	SUPERCRITICAL	LIQUID	SUPERCRITICAL	LIQUID
TECHNOLOGY (20 POINTS)				
HARDWARE TECHNOLOGY EXTENSION (10)	7	8	4	5
PROPELLANT ACQUISITION (10)	7	6	7	5
	(14)	(14)	(11)	(10)
SIMPLICITY (15 POINTS)				
NUMBER OF COMPONENTS (5)	4	5	2	3
OPERATIONAL COMPLEXITY (5)	4	5	1	2
INTEGRATION COMPLEXITY (3)	2	3	3	3
CONTROL REQUIREMENTS (2)	1	2	1	2
	(11)	(15)	(7)	(10)
FLEXIBILITY (25 POINTS)				
OPERATING CONSTRAINTS (10)	8	8	9	9
SENSITIVITY TO MAX ΔV BURN (5)	3	3	4	4
SENSITIVITY TO THRUST LEVEL (5)	2	5	2	5
SENSITIVITY TO TIME BETWEEN BURNS (3)	1	2	3	3
SENSITIVITY TO TOTAL IMPULSE (2)	1	2	1	1
	(15)	(20)	(19)	(22)
WEIGHT AND VOLUME (25 POINTS)	9	25	5	24
80 LB (1PT)	0	0	0	0
60 CU FT (1PT)	(9)	(25)	(5)	(23)
DEVELOPMENT (15 POINTS)				
ENVIRONMENTAL SIMULATION (6)	3	4	4	5
INTEGRATED TEST REQUIREMENTS (6)	3	3	5	5
FACILITY REQUIREMENTS (3)	2	2	1	3
	(8)	(9)	(10)	(13)
TOTAL	(57)	(83)√	(48)	(78)

√SELECTED

FIGURE 5-12

SYSTEM CONCEPT COMPARISON (P/T CONTROL)
o ORBITER B
$\Delta V \leq 10$ FT/SEC

SELECTION CRITERIA	PASSIVE		ACTIVE	
	SUPERCRITICAL	LIQUID	SUPERCRITICAL	LIQUID
TECHNOLOGY (20 POINTS)				
HARDWARE TECHNOLOGY EXTENSION (10)	7	8	4	5
PROPELLANT ACQUISITION (10)	4	6	4	5
	(11)	(14)	(8)	(10)
SIMPLICITY (15 POINTS)				
NUMBER OF COMPONENTS (5)	4	5	2	3
OPERATIONAL COMPLEXITY (5)	4	5	1	2
INTEGRATION COMPLEXITY (3)	2	1	3	3
CONTROL REQUIREMENTS (2)	1	1	1	2
	(11)	(12)	(7)	(10)
FLEXIBILITY (25 POINTS)				
OPERATING CONSTRAINTS (10)	6	8	7	9
SENSITIVITY TO MAX ΔV BURN (5)	3	2	4	4
SENSITIVITY TO THRUST LEVEL (5)	2	5	2	5
SENSITIVITY TO TIME BETWEEN BURNS (3)	1	1	3	3
SENSITIVITY TO TOTAL IMPULSE (2)	1	2	1	1
	(13)	(18)	(17)	(22)
WEIGHT AND VOLUME (25 POINTS)	17	25	9	25
80 LB (1PT)	0	0	0	0
60 CU FT (1 PT)	(17)	(25)	(9)	(25)
DEVELOPMENT (15 POINTS)				
ENVIRONMENTAL SIMULATION (6)	2	4	3	5
INTEGRATED TEST REQUIREMENTS (6)	2	3	4	5
FACILITY REQUIREMENTS (3)	2	2	1	3
	(6)	(9)	(8)	(13)
TOTAL POINTS	58	78	49	80√

√SELECTED

FIGURE 5-13

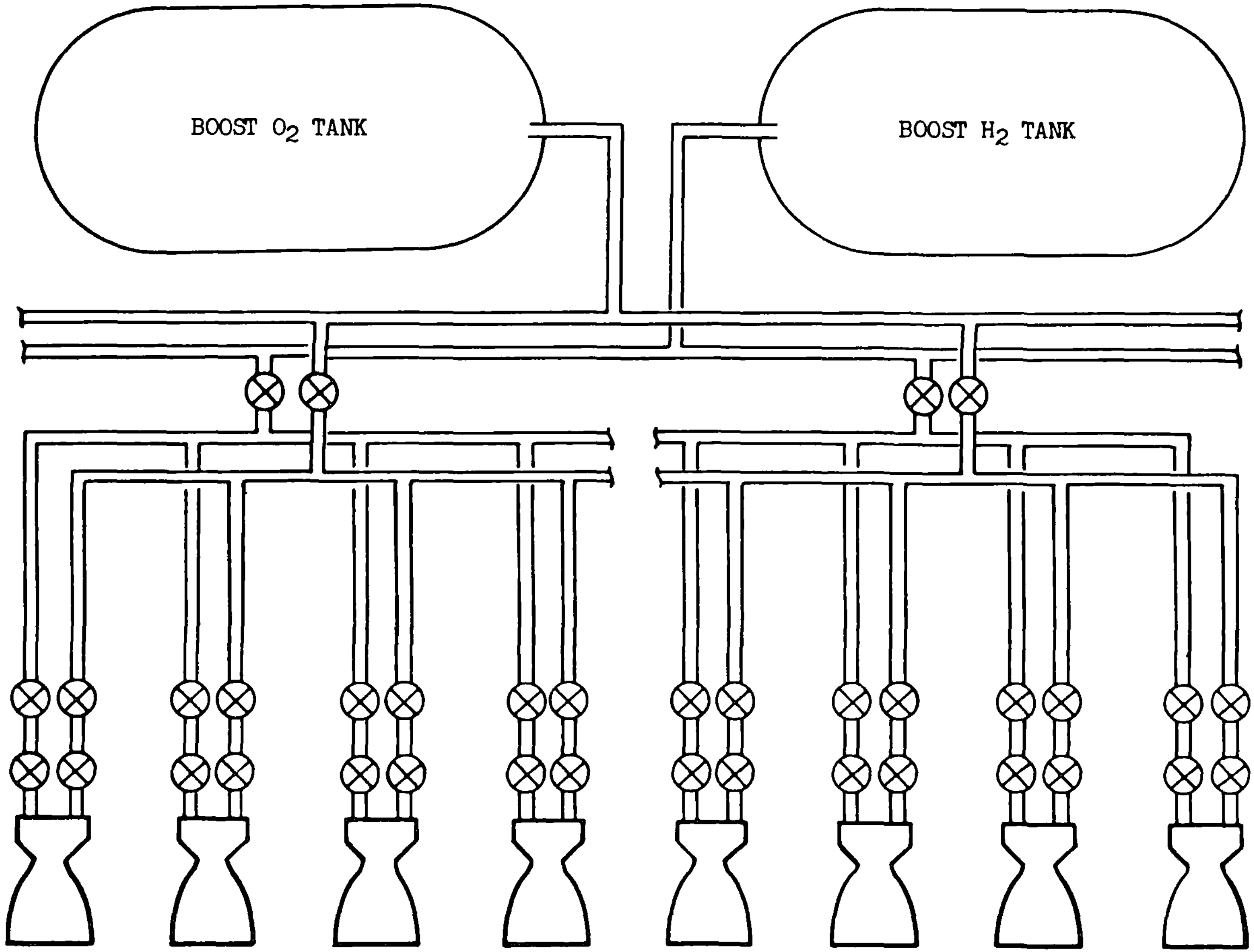
SYSTEM CONCEPT COMPARISON (P/T CONTROL)
ORBITER B
$\Delta V \leq 50$ FT/SEC

Propellant storage and thermal conditioning options were not compared for high maneuver level missions, wherein all translational maneuvers were performed by the APS, since studies of Appendix C showed a large advantage in favor of liquid storage and active thermal conditioning for this mission.

5.3 Baseline Subsystems - Based on the foregoing concept comparison and selection, recommended baseline subsystems were defined and are shown schematically in Figures 5-14 through 5-16. Component redundancy was incorporated to comply with mission failure constraints, which specify that the APS be fully operational after one component failure and provide a safe reentry after two component failures. Orbiter APS design characteristics and weight breakdowns are tabulated in Figures 5-17 through 5-22 for all three mission velocity levels. Similar data for the booster are presented in Figures 5-23. In addition, substantiation of subsystem performance is provided by Figures 5-24 through 5-31, which show histories for main engine tank pressure, engine thrust, and mixture ratio during simulated mission duty cycles. (Off-design mixture ratio and thrust excursions, which occurred in the orbiter low and intermediate velocity level missions, resulted when tank pressure decayed below the design regulator outlet pressure of 20 lb/in^2a.) Finally, each candidate orbiter subsystem was analyzed to determine sensitivity to changes in vehicle design and mission requirements. This ensured that recommended concept selections would not be invalidated by later vehicle changes. The possible changes investigated were:

1) eliminating the fail-operational reliability requirement
2) doubling the attitude acceleration requirements
3) allowing no engine penetration of the heat shield
4) reducing main engine tank pressure, and
5) eliminating main engine tank residual liquids available for APS usage.

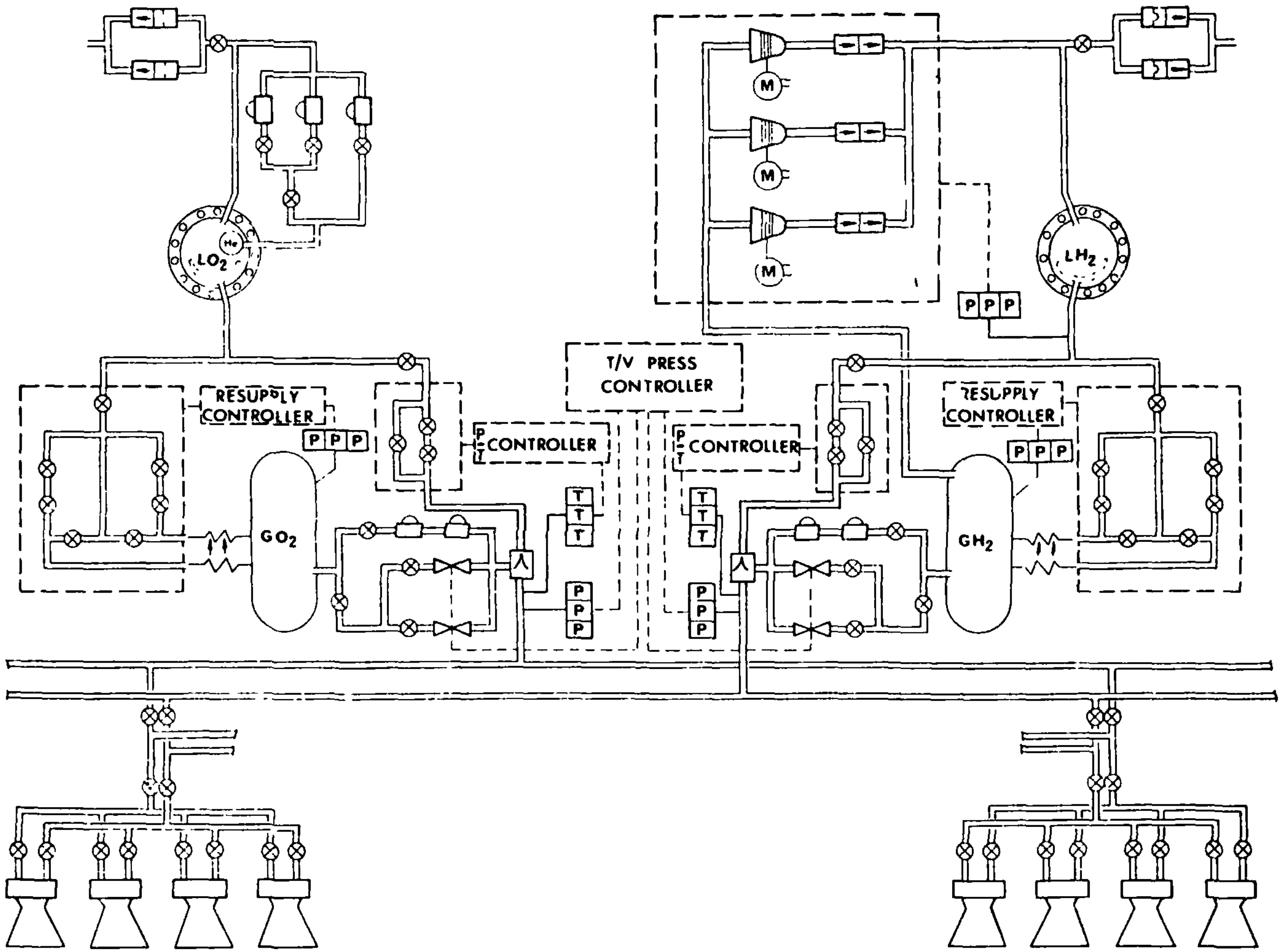
The effect of these alternate constraints on subsystem design is tabulated in Figure 5-32. Subsystem linear weight sensitivities to these alternate constraints are presented in Figures 5-33 and 5-34 for orbiters A and B, respectively. These results indicated recommended baseline selections were valid over the expected range of shuttle design and mission requirement variations.



BOOSTER A AND B SUBSYSTEM SCHEMATIC

BLOWDOWN

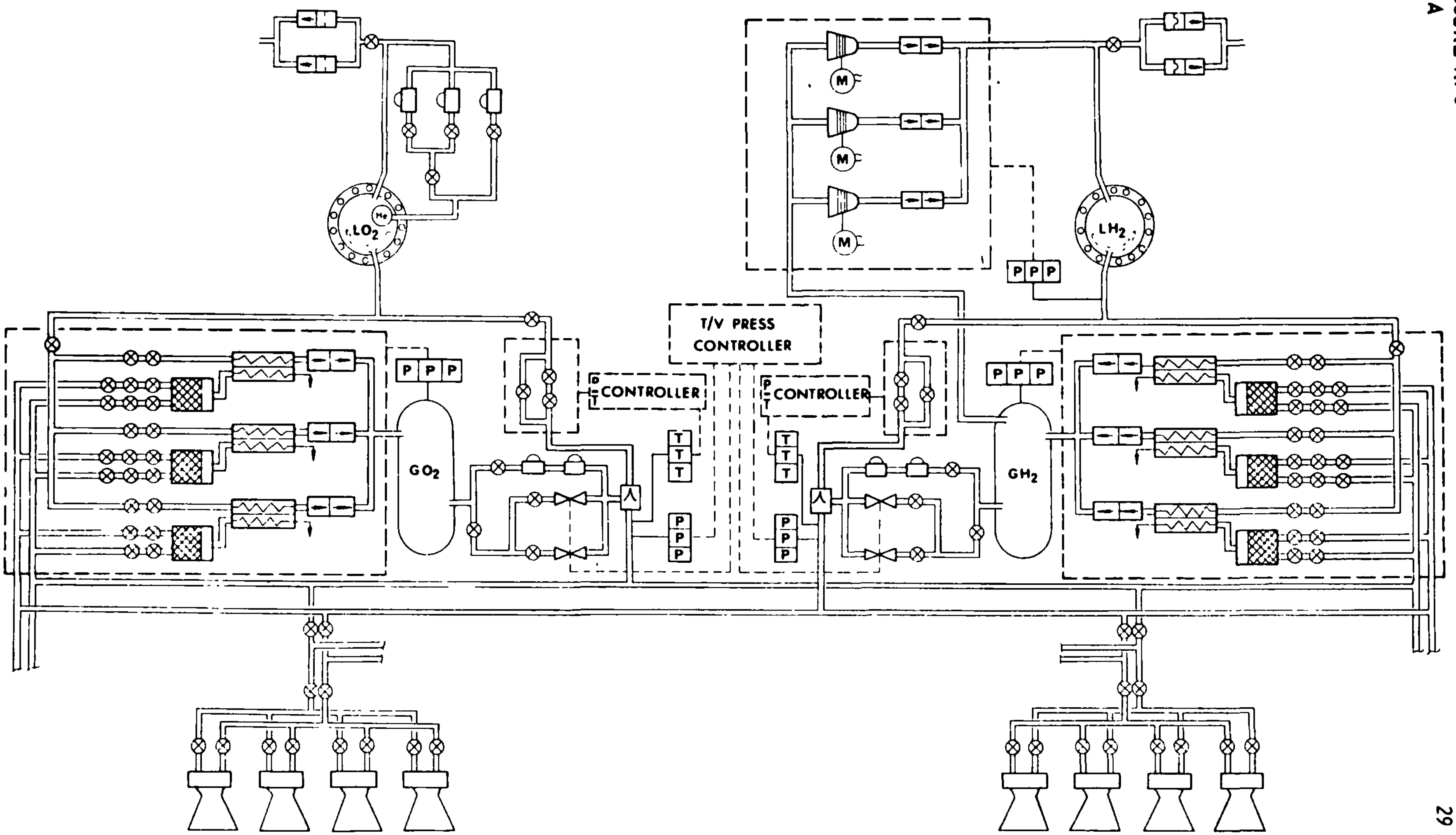
FIGURE 5-14



ORBITER A AND B SUBSYSTEM SCHEMATIC

- o LIQUID STORAGE
- o PASSIVE CONDITIONING
- o P/T CONTROL
- o $\Delta V \leq 10$ FT/SEC

FIGURE 5-15



ORBITER A AND B SUBSYSTEM SCHEMATIC

- o LIQUID STORAGE
- o ACTIVE CONDITIONING
- o P/T CONTROL
- o $\Delta V \leq 50$ FT/SEC, AND
- o ALL MANEUVER

FIGURE 5-16

	ORBITER A H_2	ORBITER A O_2	ORBITER B H_2	ORBITER B O_2
° PROPELLANT STORAGE AND PRESSURIZATION				
PROPELLANT WEIGHT, LB	802.	2406.	1037.	3112.
PROPELLANT STORAGE STATE	LIQUID	LIQUID	LIQUID	LIQUID
PRESSURIZATION TYPE	AUTOGENOUS (COMPRESSOR)	REGULATED HELIUM	AUTOGENOUS (COMPRESSOR)	REGULATED HELIUM
PROPELLANT SUPPLY PRESSURE, LB/IN^{2}A	50.	40.	45.	45.
TANK VOLUME, FT^3	193.	36.	250.	46.
° PROPELLANT THERMAL CONDITIONING				
HEAT EXCHANGER TYPE	NONE	PASSIVE	PASSIVE	PASSIVE
MAXIMUM DESIGN FLOWRATE, LB/SEC	--	4.2	1.5	6.36
OUTLET PROPELLANT TEMPERATURE, °R (AT MAXIMUM FLOWRATE)	--	176. (SAT. VAPOR)	122.	182. (50% QUALITY)
° MAIN ENGINE TANKS				
INITIAL PRESSURE, LB/IN^{2}A	43.	33.	41.	41.
INITIAL TEMPERATURE, °R	200.	300.	300.	300.
OPERATING PRESSURE RANGE, LB/IN^{2}A	43.-19.9	33.-18.4	41.-16.6	41.-16.
OPERATING TEMPERATURE RANGE, °R	199.-529.	299.-525.	527.-248	524.-233.
° PROPELLANT FLOW CONTROL				
FLOW CONTROL TYPE	CONSTANT DENSITY	CONSTANT DENSITY	CONSTANT DENSITY	CONSTANT DENSITY
GAS SIDE REGULATED PRESSURE, LB/IN^{2}A	20.	20.	20.	20.
LIQUID INLET PRESSURE, LB/IN^{2}A	30.	30.	30.	30.
LIQUID INLET TEMPERATURE, °R	SAT.LIQUID	SAT.LIQUID	SAT.LIQUID	SAT.LIQUID
MIXER OUTLET TEMPERATURE, °R	200.	300.	300.	300.
° DISTRIBUTION SUBSYSTEM AND ENGINES				
MAXIMUM FEEDLINE DIAMETER, IN	5.7	5.3	7.4	7.0
MAXIMUM FEEDLINE LENGTH, FT	69.0	69.0	124.0	124.0
MAXIMUM PROPELLANT FLOWRATE, LB/SEC	1.68	5.05	3.3	9.9
MAXIMUM NUMBER OF ENGINES FIRING	5.		5.	
DESIGN ENGINE THRUST, LB	500.		1000.	
DESIGN INJECTOR INLET PRESS., LB/IN^{2}A	15.3		15.8	
DESIGN CHAMBER PRESSURE, LB/IN^{2}A	13.3		13.8	
DESIGN MIXTURE RATIO	3.		3.	
EXPANSION RATIO	8.		8.	

ORBITER APS BASELINE DESIGN SUMMARY

° +X AXIS MANEUVERS ≤ 10 FT/SEC

FIGURE 5-17

	ORBITER A		ORBITER B	
	H_2	O_2	H_2	O_2
• PROPELLANT STORAGE AND PRESSURIZATION				
PROPELLANT WEIGHT, LB	2019.	5627.	2534.	6861.
PROPELLANT STORAGE STATE	LIQUID	LIQUID	LIQUID	LIQUID
PRESSURIZATION TYPE	AUTOGENOUS (COMPRESSOR)	REGULATED HELIUM	AUTOGENOUS (COMPRESSOR)	REGULATED HELIUM
PROPELLANT SUPPLY PRESSURE, LB/IN^2A	50.	40.	45.	45.
TANK VOLUME, FT^3	486.	84.	611.	102.
• PROPELLANT THERMAL CONDITIONING				
HEAT EXCHANGER TYPE	ACTIVE	ACTIVE	ACTIVE	ACTIVE
MAXIMUM DESIGN FLOWRATE, LB/SEC	0.95	4.2	1.9	8.8
OUTLET PROPELLANT TEMPERATURE, °R (AT MAXIMUM FLOWRATE)	100.	265.	226.	190.
• MAIN ENGINE TANKS				
INITIAL PRESSURE, LB/IN^2A	43.	33.	41.	41.
INITIAL TEMPERATURE, °R	200.	300.	300.	300.
OPERATING PRESSURE RANGE, LB/IN^2A	43.-17.0	33.-19.2	41.-17.2	41.-17.6
OPERATING TEMPERATURE RANGE, °R	200.-528.	299.-524.	259.-526.	265.-523.
• PROPELLANT FLOW CONTROL				
FLOW CONTROL TYPE	CONSTANT DENSITY	CONSTANT DENSITY	CONSTANT DENSITY	CONSTANT DENSITY
GAS SIDE REGULATED PRESSURE, LB/IN^2A	20.	20.	20.	20.
LIQUID INLET PRESSURE, LB/IN^2A	30.	30.	30.	30.
LIQUID INLET TEMPERATURE, °R	SAT. LIQUID	SAT. LIQUID	SAT. LIQUID	SAT. LIQUID
MIXER OUTLET TEMPERATURE, °R	200.	300.	300.	300.
• DISTRIBUTION SUBSYSTEM AND ENGINES				
MAXIMUM FEEDLINE DIAMETER, IN	5.5	5.2	7.4	7.0
MAXIMUM FEEDLINE LENGTH, FT	69.	69	124	124
MAXIMUM PROPELLANT FLOWRATE, LB/SEC	1.67	5.0	3.3	9.9
MAXIMUM NUMBER OF ENGINES FIRING	5.		5.	
DESIGN ENGINE THRUST, LB	500.		1000.	
DESIGN INJECTOR INLET PRESS., LB/IN^2A	15.4		15.9	
DESIGN CHAMBER PRESSURE, LB/IN^2A	13.4		13.9	
DESIGN MIXTURE RATIO	3.		3.	
EXPANSION RATIO	10.		10.	

ORBITER APS BASELINE DESIGN SUMMARY

° +X AXIS MANEUVERS ≤ 50 FT/SEC

FIGURE 5-18

	ORBITER A H_2	ORBITER A O_2	ORBITER B H_2	ORBITER B O_2
• PROPELLANT STORAGE AND PRESSURIZATION				
PROPELLANT WEIGHT, LB	10643.	28029.	12526.	32538.
PROPELLANT STORAGE STATE	LIQUID	LIQUID	LIQUID	LIQUID
PRESSURIZATION TYPE	AUTOGENOUS (COMPRESSOR)	REGULATED HELIUM	AUTOGENOUS (COMPRESSOR)	REGULATED HELIUM
PROPELLANT SUPPLY PRESSURE, LB/IN^2A	50.	40.	45.	45.
TANK VOLUME, FT^3	2565.	418.	3019.	485.
• PROPELLANT THERMAL CONDITIONING				
HEAT EXCHANGER TYPE	ACTIVE	ACTIVE	ACTIVE	ACTIVE
MAXIMUM DESIGN FLOWRATE, LB/SEC	1.2	4.3	2.2	9.5
OUTLET PROPELLANT TEMPERATURE, °R (AT MAXIMUM FLOWRATE)	236.	328.	288.	266.
• MAIN ENGINE TANKS				
INITIAL PRESSURE, LB/IN^2A	43.	33.	41.	41.
INITIAL TEMPERATURE, °R	200.	300.	300.	300.
OPERATING PRESSURE RANGE, LB/IN^2A	43.-20.3	33.-20.5	41.-19.9	41.-19.8
OPERATING TEMPERATURE RANGE, °R	200.-529.	200.-529.	296.-526.	290.-523.
• PROPELLANT FLOW CONTROL				
FLOW CONTROL TYPE	CONSTANT DENSITY	CONSTANT DENSITY	CONSTANT DENSITY	CONSTANT DENSITY
GAS SIDE REGULATED PRESSURE, LB/IN^2A	20.	20.	20.	20.
LIQUID INLET PRESSURE, LB/IN^2A	30.	30.	30.	30.
LIQUID INLET TEMPERATURE, °R	SAT.LIQUID	SAT.LIQUID	SAT.LIQUID	SAT.LIQUID
MIXER OUTLET TEMPERATURE, °R	200.	300.	300.	300.
• DISTRIBUTION SUBSYSTEM AND ENGINES				
MAXIMUM FEEDLINE DIAMETER, IN	5.52	5.15	7.4	7.0
MAXIMUM FEEDLINE LENGTH, FT	69.0	69.0	124.0	124.0
MAXIMUM PROPELLANT FLOWRATE, LB/SEC	1.67	5.00	3.3	9.9
MAXIMUM NUMBER OF ENGINES FIRING	5.		5.	
DESIGN ENGINE THRUST, LB	500.		1000.	
DESIGN INJECTOR INLET PRESS., LB/IN^2A	15.4		15.9	
DESIGN CHAMBER PRESSURE, LB/IN^2A	13.4		13.9	
DESIGN MIXTURE RATIO	3.		3.	
EXPANSION RATIO	10.		10.	

FIGURE 5-19

ORBITER APS BASELINE DESIGN SUMMARY

° ALL MANEUVERS

APS WEIGHT	ORBITER A	ORBITER B
PROPELLANT		
HYDROGEN (H_2)	802	1037
OXYGEN (O_2)	2406	3112
HYDROGEN TANK		
PRESSURIZATION SYSTEM	122	128
TANK AND INSULATION	208	249
SCREEN	38	47
OXYGEN TANK		
PRESSURIZATION SYSTEM	17	23
TANK AND INSULATION	105	120
SCREEN	13	15
THERMAL CONDITIONING		
GAS GENERATOR (H_2)		
GAS GENERATOR (O_2)		
HEAT EXCHANGER (H_2)		173
HEAT EXCHANGER (O_2)	280	413
DISTRIBUTION SYSTEM		
LINES	406	450
CONTROLS (VALVES & REGS)	662	875
ENGINES	1613	2092
TOTAL	6672	8734

SUBSYSTEM WEIGHT SUMMARY

- CONSTANT DENSITY FLOW CONTROL
- LIQUID STORAGE/PASSIVE CONDITIONING
- ≤ 10 FPS

FIGURE 5-20

APS WEIGHT	ORBITER A	ORBITER B
PROPELLANT		
HYDROGEN (H_2)	2019	2534
OXYGEN (O_2)	5627	6861
HYDROGEN TANK		
PRESSURIZATION SYSTEM	146	154
TANK AND INSULATION	421	487
SCREEN	74	86
OXYGEN TANK		
PRESSURIZATION SYSTEM	37	50
TANK AND INSULATION	167	187
SCREEN	22	25
THERMAL CONDITIONING		
GAS GENERATOR (H_2) / GAS GENERATOR (O_2)	45	53
HEAT EXCHANGER (H_2)	128	269
HEAT EXCHANGER (O_2)	255	277
DISTRIBUTION SYSTEM		
LINES	405	451
CONTROLS (VALVES & REGS)	686	910
ENGINES	1703	2206
TOTAL	11735	14550

SUBSYSTEM WEIGHT SUMMARY

- CONSTANT DENSITY FLOW CONTROL
- LIQUID STORAGE/ACTIVE CONDITIONING
- $\leq$50 FPS

FIGURE 5-21

APS WEIGHT	ORBITER A	ORBITER B
PROPELLANT		
HYDROGEN (H_2)	10643	12526
OXYGEN (O_2)	28029	32538
HYDROGEN TANK		
PRESSURIZATION SYSTEM	252	235
TANK AND INSULATION	1850	2016
SCREEN	220	245
OXYGEN TANK		
PRESSURIZATION SYSTEM	180	234
TANK AND INSULATION	422	464
SCREEN	65	72
THERMAL CONDITIONING		
GAS GENERATOR (H_2)	49	59
GAS GENERATOR (O_2)		
HEAT EXCHANGER (H_2)	178	340
HEAT EXCHANGER (O_2)	278	326
DISTRIBUTION SYSTEM		
LINES	405	451
CONTROLS (VALVES & REGS)	686	910
ENGINES	1703	2206
TOTAL	44960	52622

SUBSYSTEM WEIGHT SUMMARY

- CONSTANT DENSITY FLOW CONTROL
- LIQUID STORAGE/ACTIVE CONDITIONING
- ALL MANEUVERS

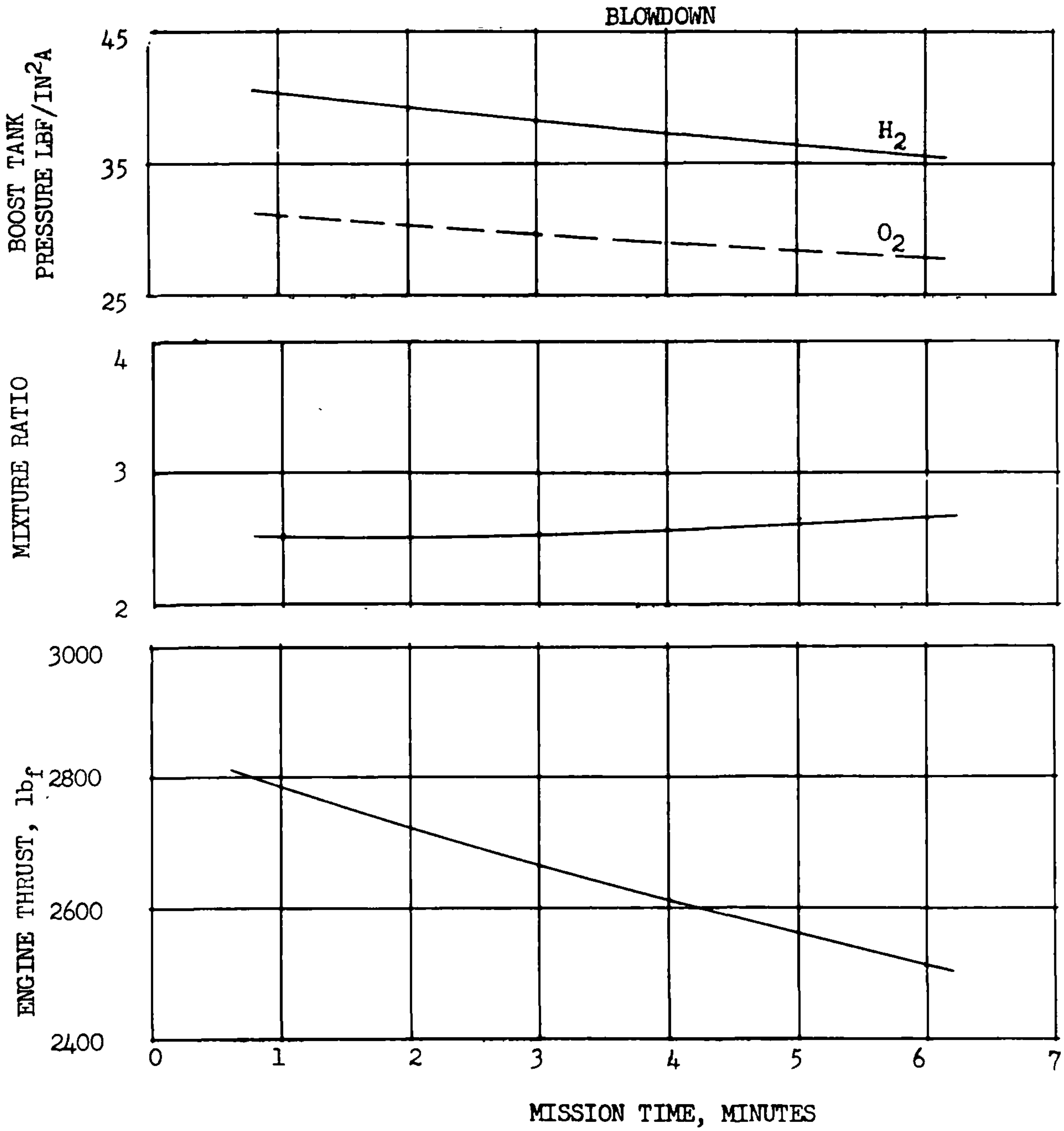
FIGURE 5-22

	BOOSTER A H_2	BOOSTER A O_2	BOOSTER B H_2	BOOSTER B O_2
° MAIN ENGINE TANKS				
INITIAL PRESSURE, LB/IN^2A	43.	33.	41.	26.
INITIAL TEMPERATURE, °R	200.	300.	300.	300.
OPERATING PRESSURE RANGE, LB/IN^2A	43.-36.2	33.-27.9	41.-33.4	26.-21.3
OPERATING TEMPERATURE RANGE, °R	200.-210.	300.-310	300.-311.	300.-310.
° DISTRIBUTION SUBSYSTEM AND ENGINES				
MAXIMUM FEEDLINE DIAMETER, IN	7.1	7.5	6.9	7.2
MAXIMUM FEEDLINE LENGTH, FT	76.0	76.0	85.0	85.0
MAXIMUM PROPELLANT FLOWRATE, LB/SEC	8.1	32.4	6.3	25.2
MAXIMUM NUMBER OF ENGINES FIRING	5.		5.	
DESIGN ENGINE THRUST, LB	2600.		2000.	
DESIGN INJECTOR INLET PRESSURE, LB/IN^2A	24.		16.	
DESIGN INJECTOR INLET TEMPERATURE, °R	200.	300.	300.	300.
DESIGN CHAMBER PRESSURE, LB/IN^2A	22.		14.	
DESIGN MIXTURE RATIO	4.		4.	
EXPANSION RATIO	2.		2.	
° SUBSYSTEM WEIGHT				
ISOLATION VALVES, LB	367.	392.	324.	348.
FEEDLINES, LB	297.	328.	384.	422.
ENGINE ASSEMBLIES, LB	1588.		1611.	
OVERALL SUBSYSTEM WEIGHT, LB	2972.		3089.	

BOOSTER APS BASELINE DESIGN SUMMARY

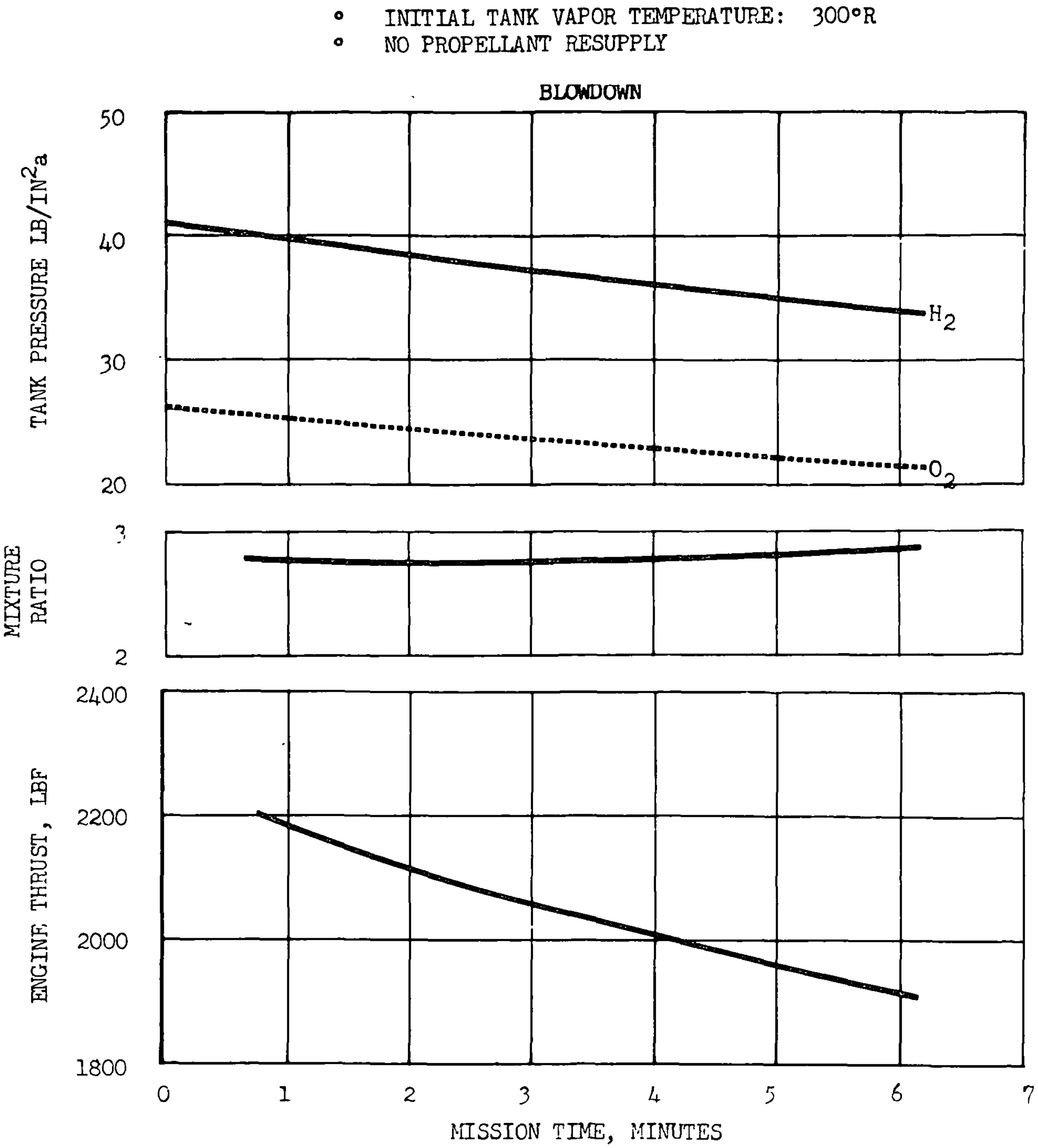
FIGURE 5-23

- INITIAL TANK VAPOR TEMPERATURES: H_2 = 200°R O_2 = 300°R
- NO PROPELLANT RESUPPLY



VEHICLE A BOOSTER APS OPERATIONAL CHARACTERISTICS

FIGURE 5-24



VEHICLE B BOOSTER APS OPERATIONAL CHARACTERISTICS

FIGURE 5-25

- CONSTANT DENSITY PROPELLANT FLOW CONTROL
- +X AXIS MANEUVERS $\leq$ 10 FT/SEC
- PASSIVE THERMAL CONDITIONING OF O_2 ONLY
- RESUPPLY PROPELLANT TEMPERATURES
  - H_2 = 45°R (SATURATED LIQUID)
  - O_2 = 182°R (SATURATED VAPOR)

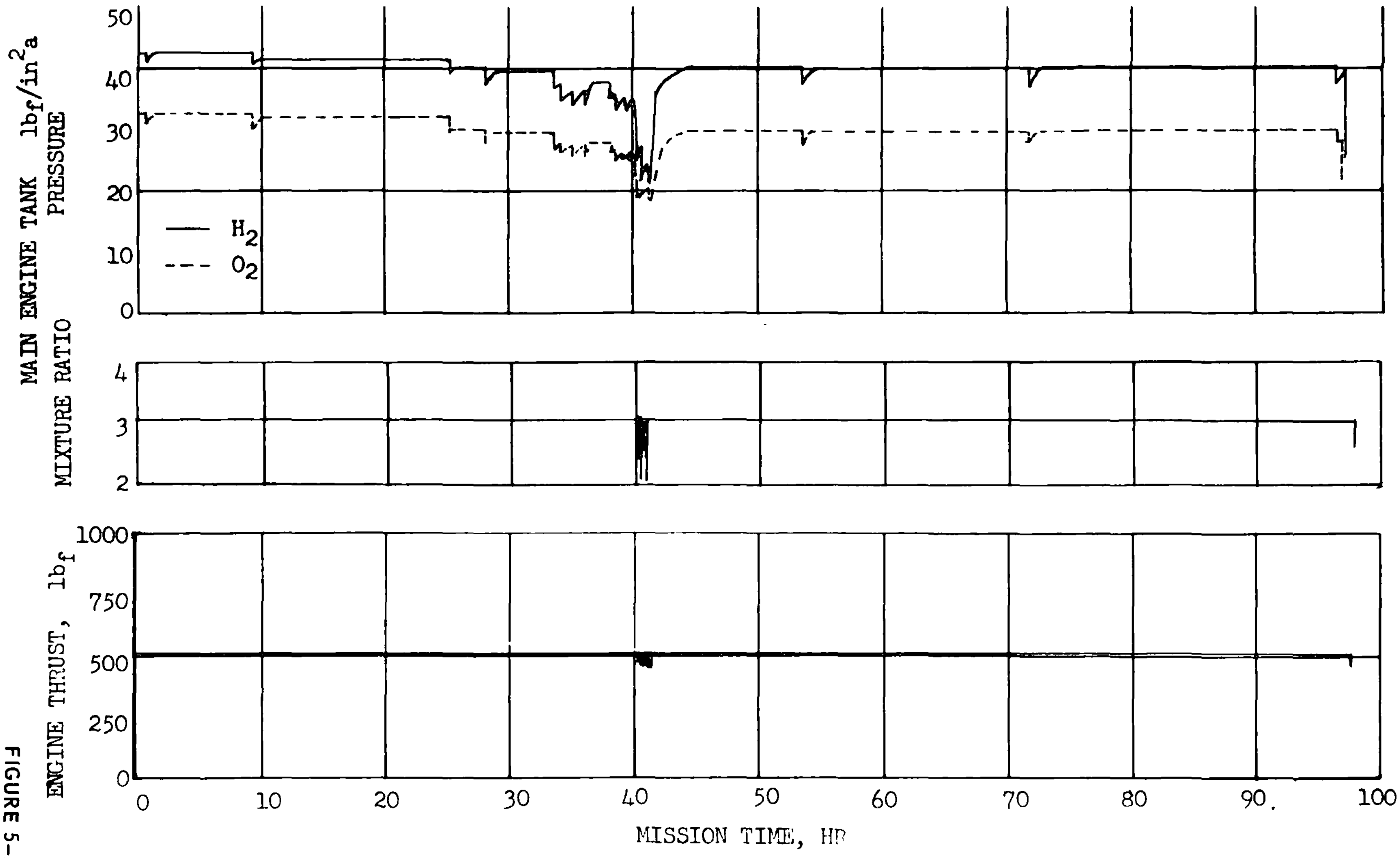


FIGURE 5-26
VEHICLE A ORBITER SIMULATED APS MISSION

- CONSTANT DENSITY PROPELLANT FLOW CONTROL
- +X AXIS MANEUVERS $\leq$ 50 FT/SEC
- ACTIVE THERMAL CONDITIONING
- RESUPPLY PROPELLANT TEMPERATURES
  H_2 = 100°R
  O_2 = 265°R

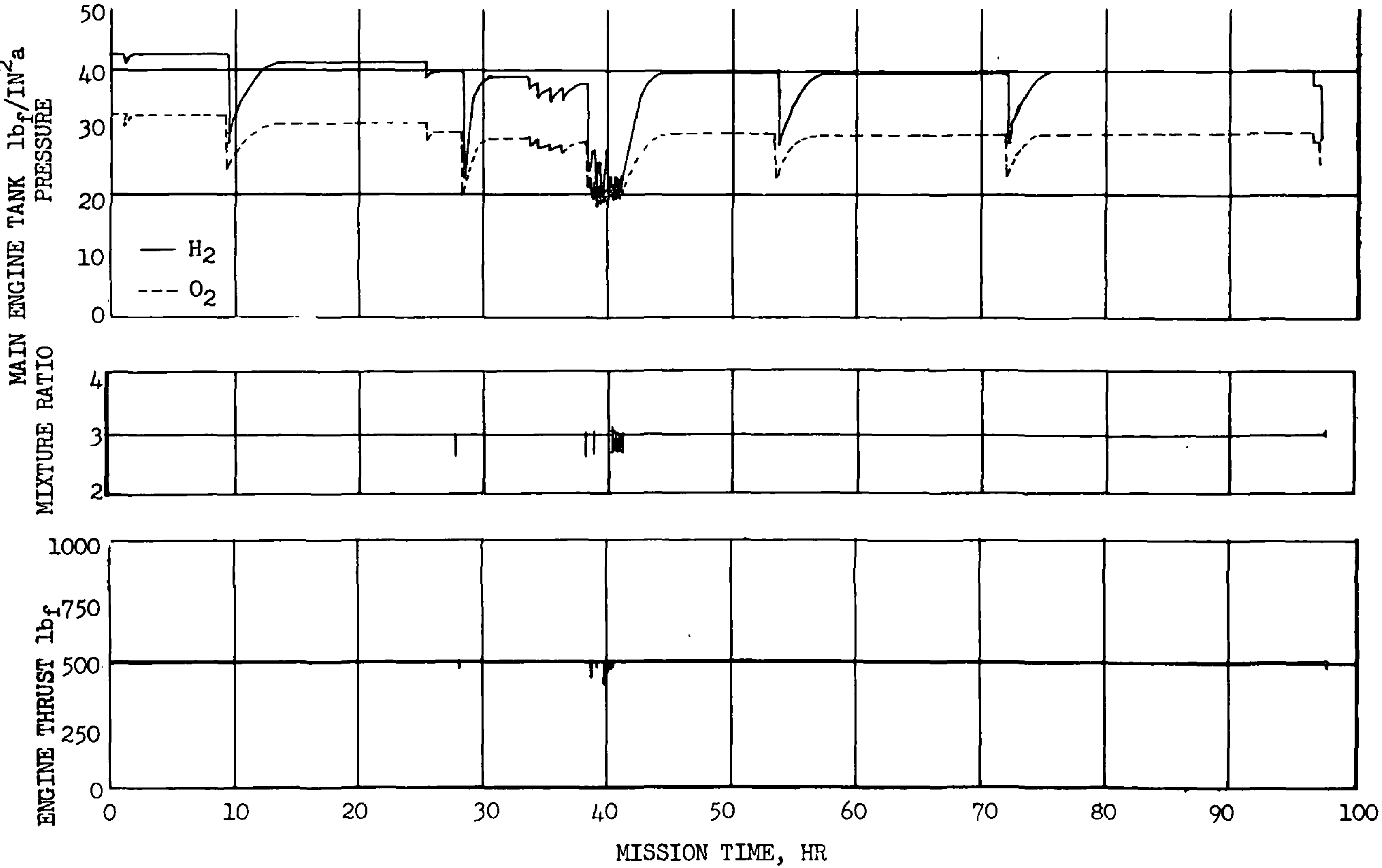


FIGURE 5-27
VEHICLE A ORBITER SIMULATED APS MISSION

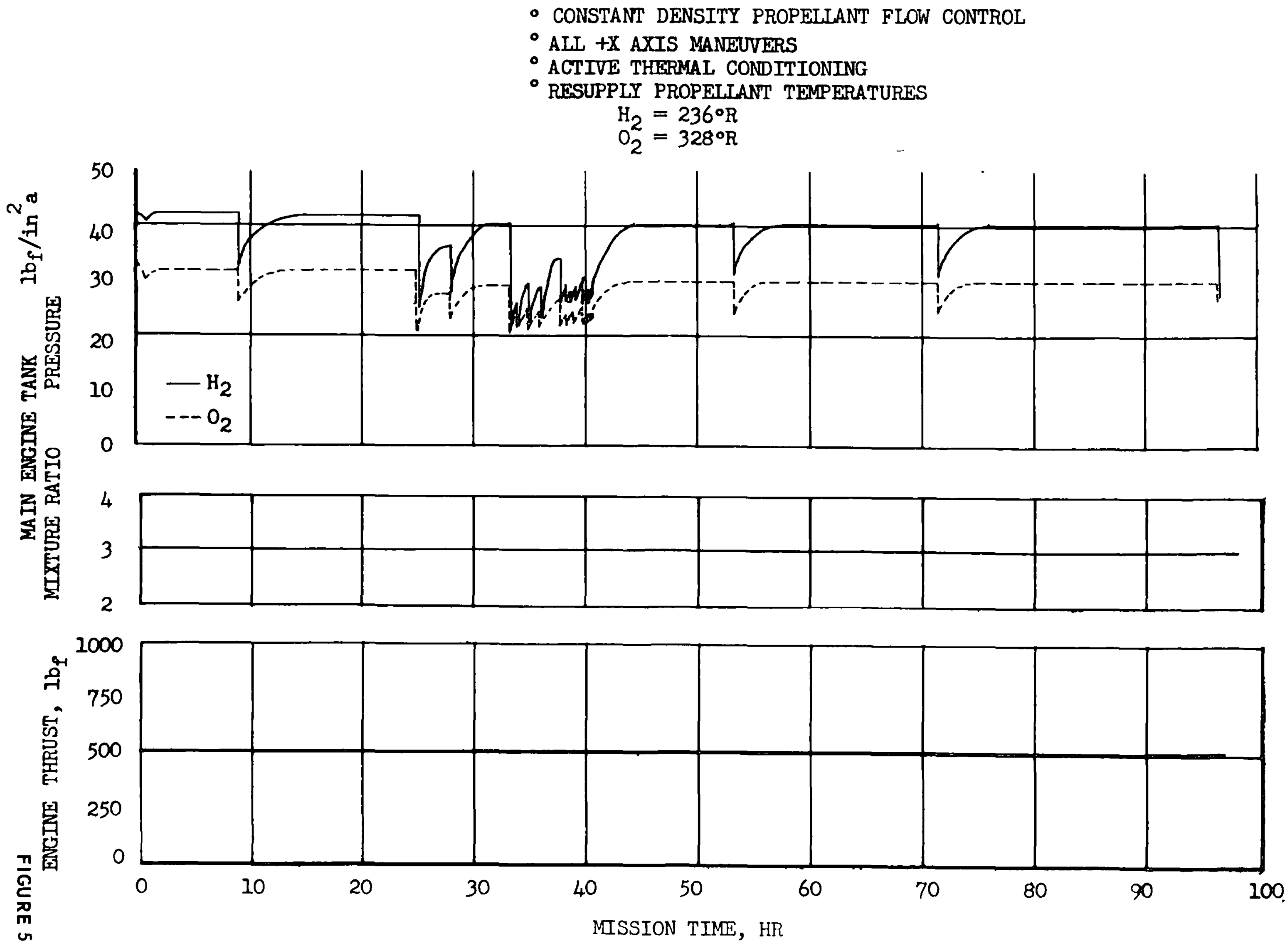


FIGURE 5-28
VEHICLE A ORBITER SIMULATED APS MISSION

- CONSTANT DENSITY PROPELLANT FLOW CONTROL
- +X AXIS MANEUVERS ≤10 FT/SEC
- PASSIVE THERMAL CONDITIONING
- RESUPPLY PROPELLANT TEMPERATURES
- H_2 = 122°R
  O_2 = 181°R (50% QUALITY)

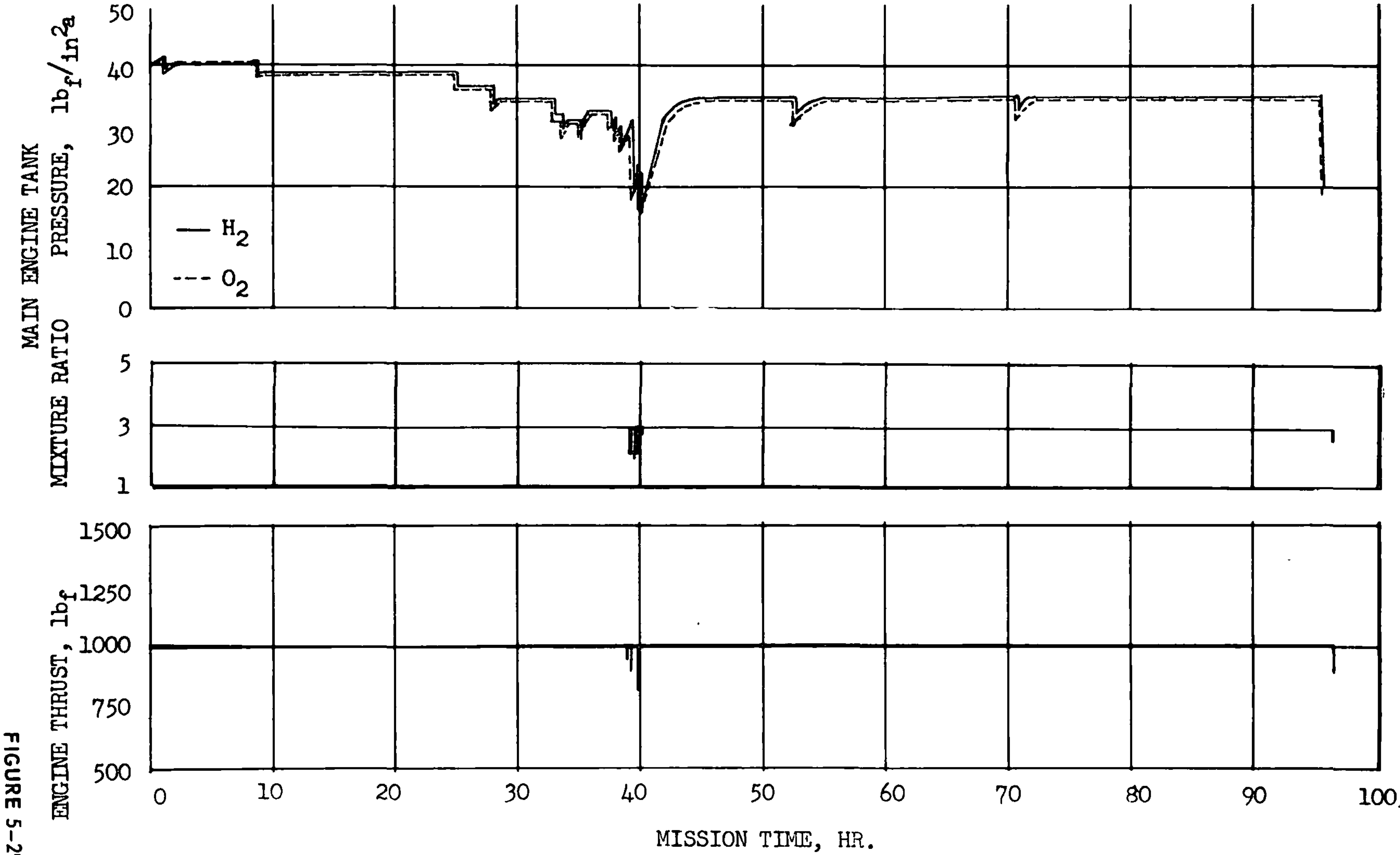


FIGURE 5-29

VEHICLE B ORBITER SIMULATED APS MISSION

- CONSTANT DENSITY PROPELLANT FLOW CONTROL
- +X AXIS MANEUVERS ≤ 50 FT/SEC
- ACTIVE THERMAL CONDITIONING
- RESUPPLY PROPELLANT TEMPERATURES
  H_2 = 226°R
  O_2 = 190°R

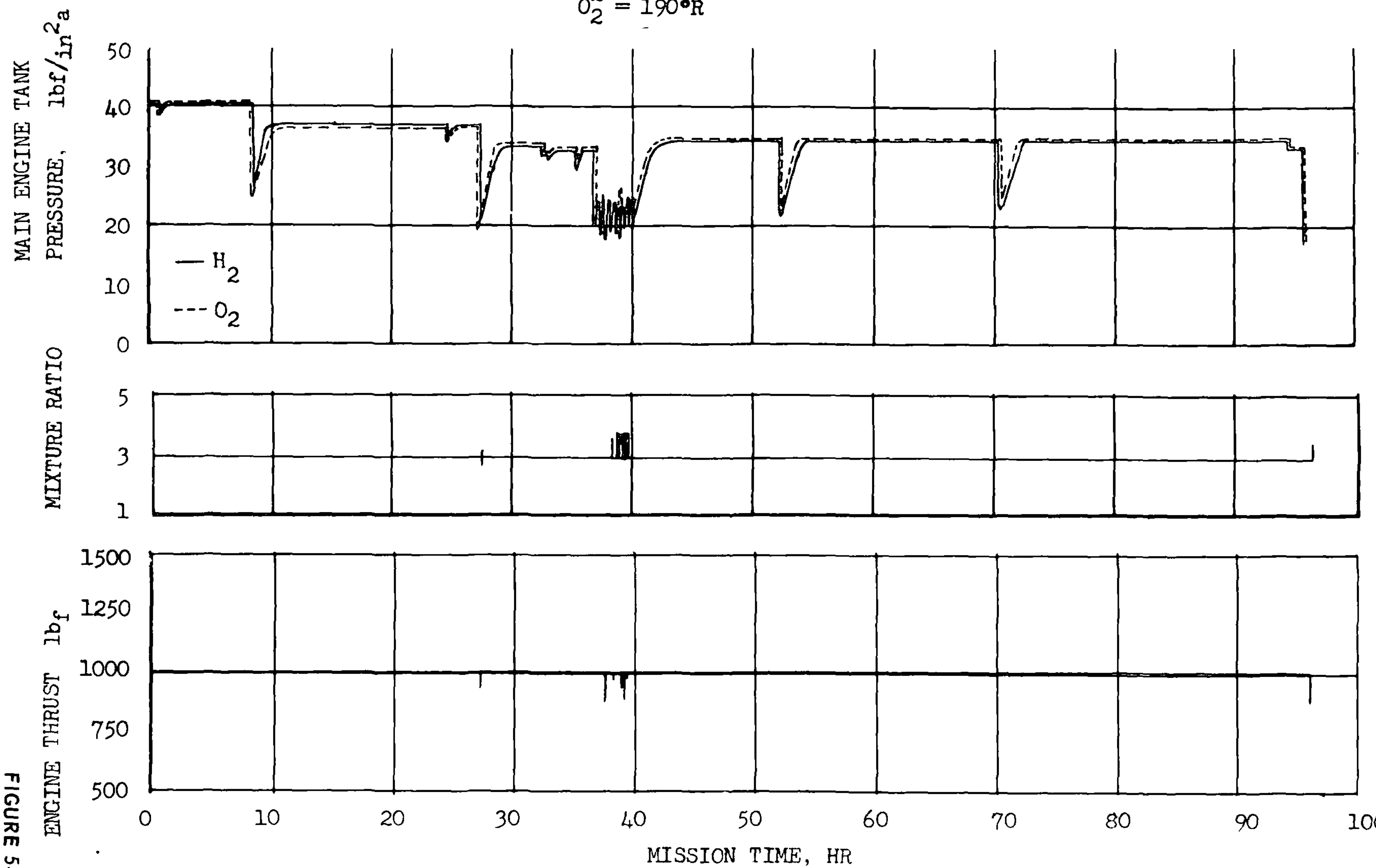


FIGURE 5-30

VEHICLE B ORBITER SIMULATED APS MISSION

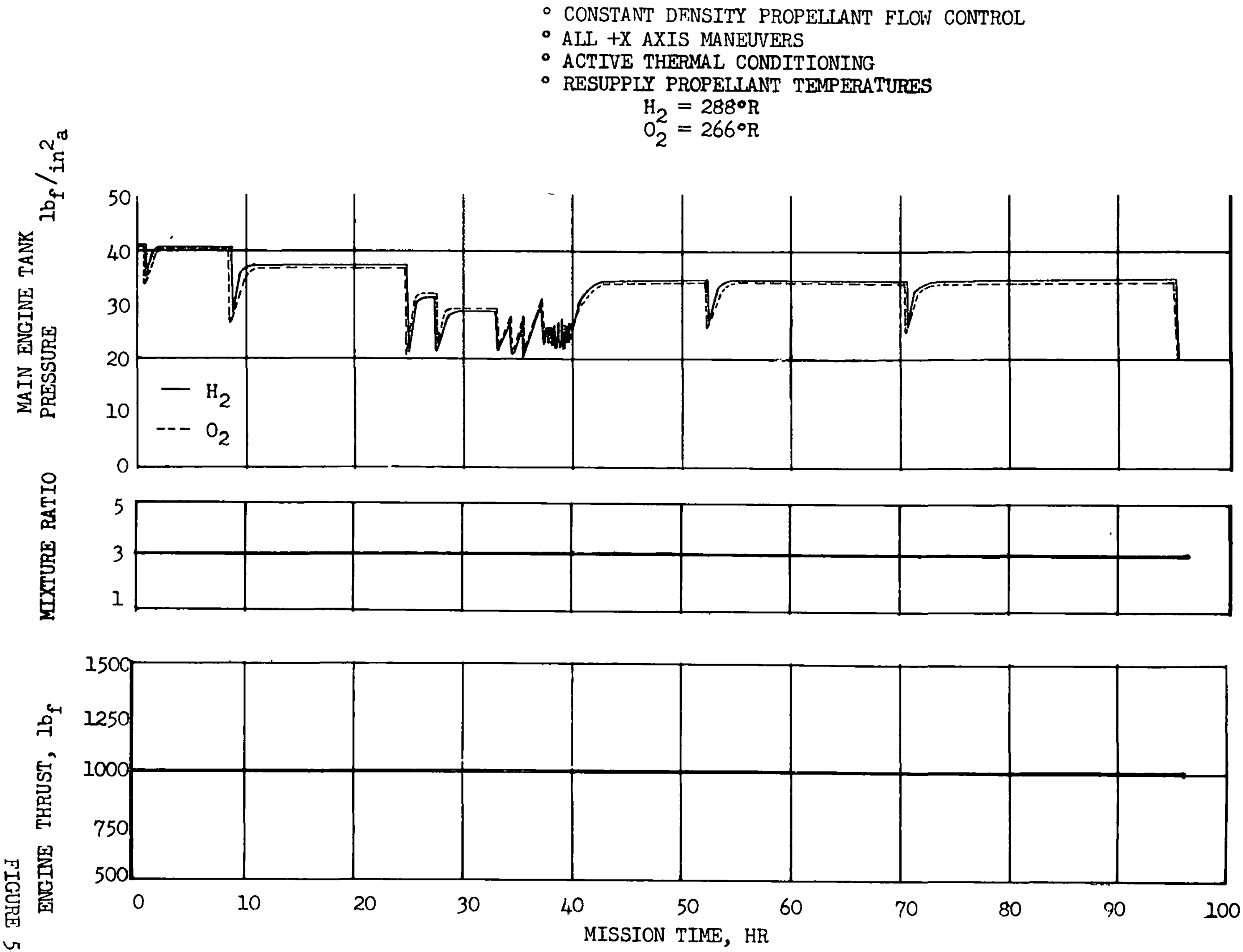


FIGURE 5-31

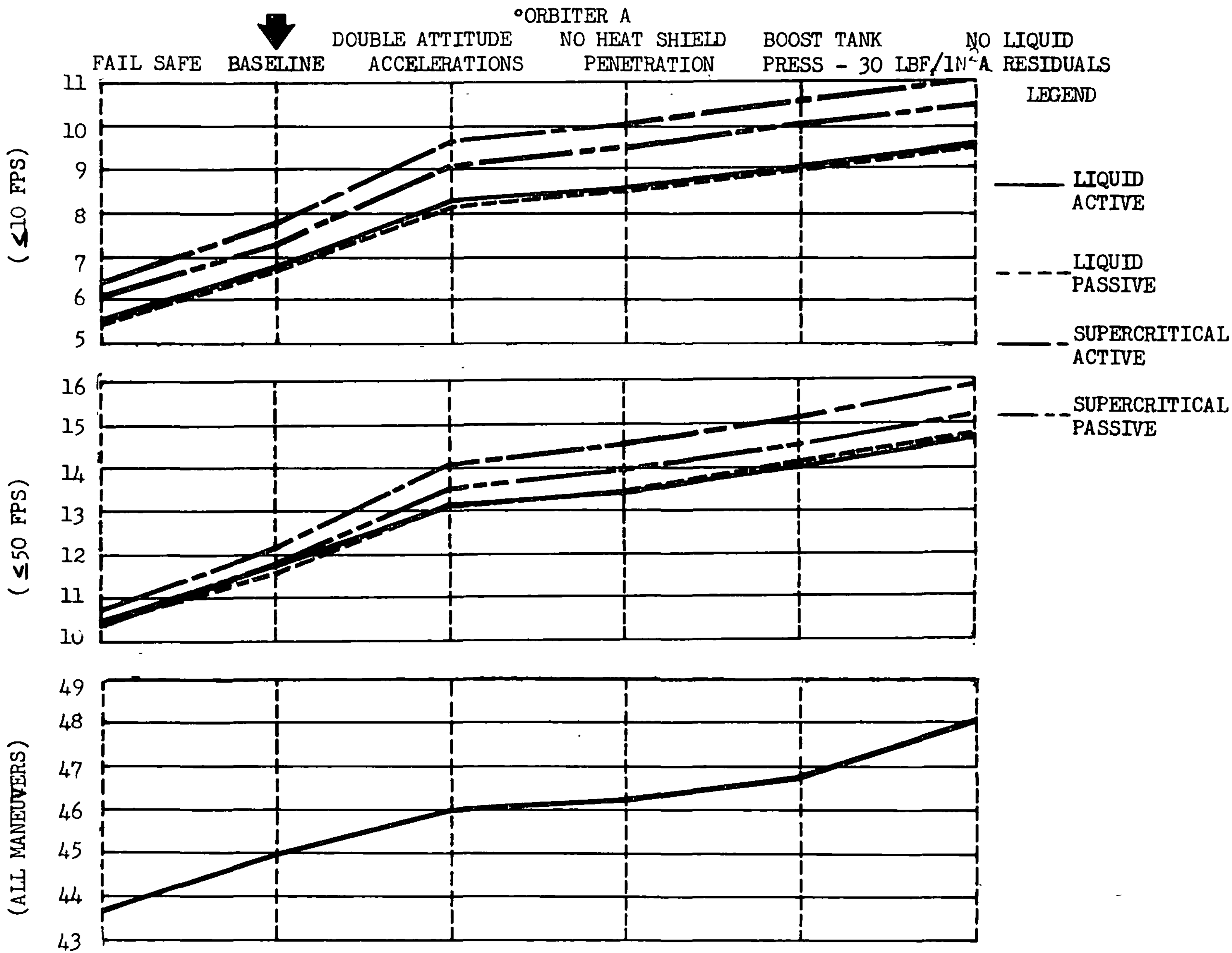
VEHICLE B ORBITER SIMULATED APS MISSION

- ORBITER A AND B
- P/T CONTROL

DESIGN CONSTRAINT	ORBITER A		ORBITER B	
	BASE	VAR	BASE	VAR
FAIL SAFE				
NO. OF THRUSTERS	32	16	28	14
NO. OF VALVES & REGS. (LIQ/PAS)	62	32	56	30
NO. OF THERMAL COND. SYSTEMS	3	2	3	2
NO. OF PRESSURIZATION SYSTEMS	3	2	3	2
DOUBLE ATT. ACCELERATION				
THRUST LEVEL, LBS	500	1000	1000	2000
NO HEAT SHIELD PENETRATION				
NO. OF THRUSTERS	32	37	28	32
Δ IMPULSE, LB-SEC	-	40500	-	55900
BOOST TANK PRESSURE				
O_2 MAXIMUM VENT PRESSURE, PSIA	35	30	46	41
OPTIMUM CHAMBER PRESSURE, PSIA	13.3	9.4	13.9	9.6
NO LIQUID RESIDUALS				
Δ IMPULSE, LB-SECS $\leq$10 FPS	-	134000	-	28300
Δ IMPULSE, LB-SECS $\leq$50 FPS	-	198000	-	54400
Δ IMPULSE, LB-SECS ALL	-	378000	-	63800

FIGURE 5-32

EFFECT OF ALTERNATE DESIGN CONSTRAINTS ON SUBSYSTEM DESIGN



SUBSYSTEM SENSITIVITIES TO DESIGN CONSTRAINTS

FIGURE 5-33

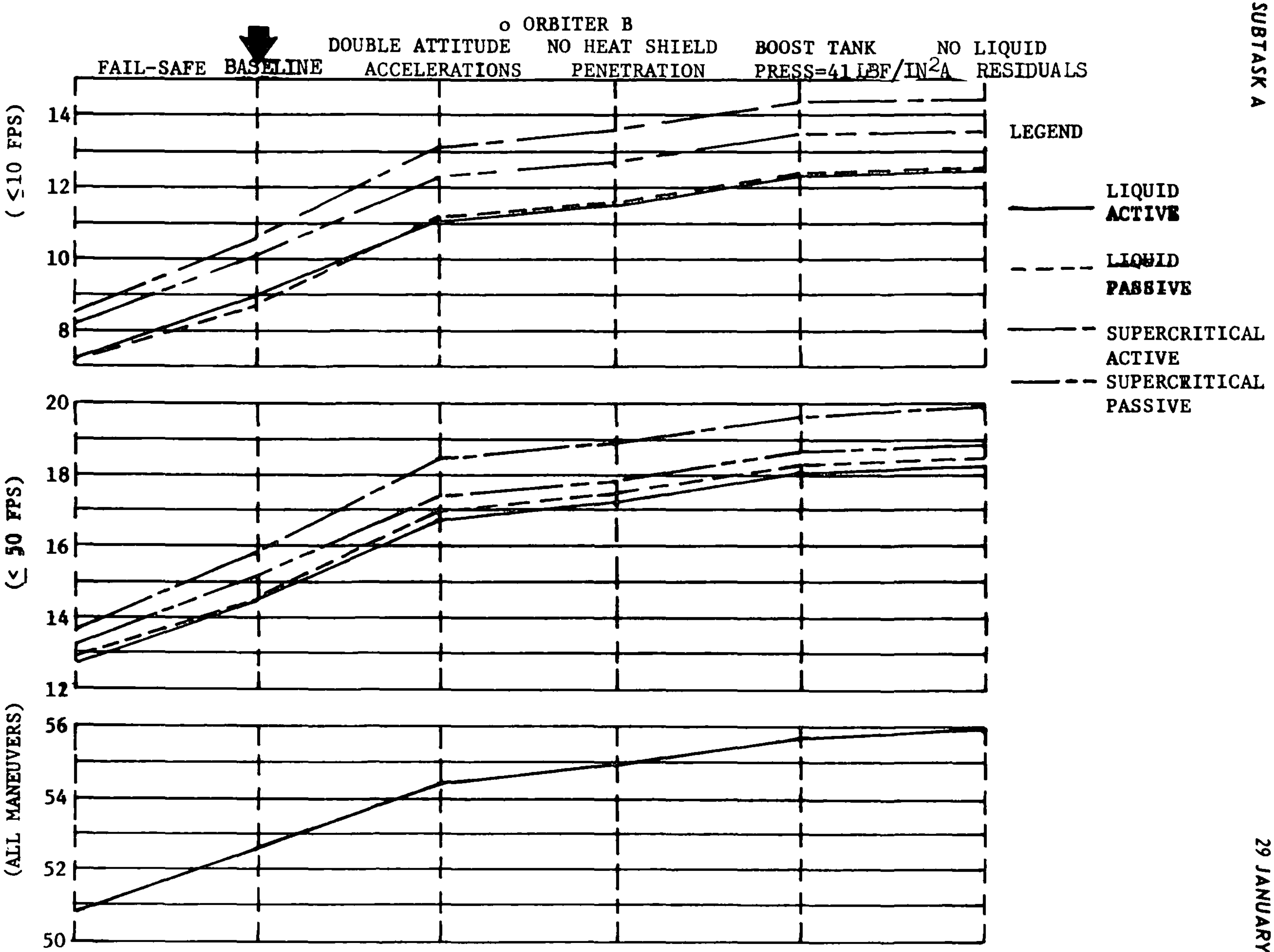


FIGURE 5-34

SUBSYSTEM SENSITIVITIES TO DESIGN CONSTRAINTS

6. CONCLUSIONS AND RECOMMENDATIONS

Numerous conceptual low pressure APS candidates were evaluated for the two space shuttle configurations defined in Appendix A. The number of potential candidates was reduced through prerequisite screening studies, and the remaining concepts compared by applying a common set of criteria consistent with space shuttle goals and constraints. This resulted in identification of recommended baseline subsystems for each of two boosters and orbiters. Only one mission timeline was considered for the booster vehicles, while three distinct timelines were considered for the orbiters, depending upon the number of +X maneuvers performed by the APS.

The most important result of booster analysis was that mission total impulse requirements could be satisfied entirely using residual gaseous propellants contained in the main engine tanks following boost. This eliminated the requirement for both a propellant storage and a propellant thermal conditioning assembly. In addition, it was found that satisfactory engine performance could be achieved by allowing the subsystem to operate in a blowdown mode without exercising any control over engine injector inlet pressure or temperature.

For orbiter missions it was found that additional propellant (separate from the residual propellant contained in the main engine tanks) was required to satisfy APS impulse demands. Storage of this additional propellant as a liquid in separate, nonrefillable tanks provided the simplest design and was considered superior to supercritical storage or integral storage with OMS propellant. Moreover, with only one exception, it was found that thermal conditioning of propellant supplied to the main engine tanks was necessary to maintain tank pressure and temperature above the minimum limits required for reliable engine operation. The exception was the minimum ΔV mission (i.e., +X Axis maneuvers ≤ 10 ft/sec) for shuttle A orbiter, where it was found that injecting unconditioned liquid hydrogen into the main engine tank maintained acceptable pressures and temperatures. In all other minimum ΔV mission cases, propellant thermal conditioning was best achieved by using passive heat sink heat exchangers. However, due to prohibitive vehicle surface area requirements for the passive heat exchanger, an active heat exchanger/gas generator thermal conditioning assembly was preferred for maximum ΔV mission (in which all vehicle maneuvers are performed by the APS). For the intermediate ΔV mission (i.e., +X axis maneuvers ≤ 50 ft/sec) the passive and active heat exchanger concepts were equally competitive. The active approach was rated slightly more attractive, however, because of its weight advantage, reduced environmental sensitivity, and nonintegrated subsystem test requirements.

Minimum orbiter APS weight and maximum mission flexibility were achieved by supplying propellant to the engines at constant density, enabling the engines to operate at constant thrust, and mixture ratio during major burns. This was achieved by employing a liquid/vapor mixing chamber in the main propellant feedline where liquid propellant was mixed with gaseous propellant withdrawn from the main engine tanks. Both liquid and vapor flows were regulated to constant pressure prior to mixing, and liquid flowrate was throttled as required to maintain constant mixer outlet temperature. This approach allowed the engine to operate at optimum design conditions for most of the mission, and reduced thermal conditioning assembly flow and energy requirements.

Primary technology issues pertaining to low pressure APS are summarized in Figures 6-1 through 6-6, and were recommended for additional in-depth study during Subtask B. With regard to engine assemblies major emphasis should be placed on minimizing chamber and valve weight. Additional design studies are required for propellant storage to ensure performance of tank insulation following repeated vent and repressurization cycles, and to devise a surface tension screen propellant acquisition device which can be tested in a 1 g environment. In the area of propellant thermal conditioning, the optimal location for a passive heat-sink type heat exchanger requires better definition (i.e., vehicle fuselage, wings, or main engine propellant tanks). The efficiency with which heat may be delivered from a passive heat exchanger to the propellant fluid should be established, since not all of the incident radiant energy or sink material internal energy can be usefully employed. Also pertinent to the study of propellant thermal conditioning are main engine tank thermodynamics. Alternate schemes should be investigated to achieve greater use of residual liquid propellant, and nonequilibrium effects such as propellant stratification should be studied to determine their effect on thermal conditioning assembly flow and energy requirements. Finally, in the area of propellant flow control, mixing chamber design requirements (i.e., injector orifice size, spray pattern, mixing length) and control requirements (liquid throttling, regulator location, and instrumentation) should be determined.

DESIGN POINT EVALUATION

- SPECIFIED OPERATING REQUIREMENTS WITHIN FUNCTIONAL LIMITS
- THRUSTER PERFORMANCE BASED ON DEMONSTRATED INJECTOR
- MINIMUM IMPULSE BIT OF 15 LB-SEC REQUIRES FAST PROPELLANT VALVES

KEY TECHNOLOGY AREAS

- THRUSTER DESIGN SHOULD STRESS MINIMUM WEIGHT
  - LIGHTWEIGHT CONSTRUCTION MATERIALS; BIMETAL JOINTS
  - MINIMIZE INJECTOR DIAMETER THROUGH COMBUSTION CHAMBER CONTRACTION RATIO TRADEOFF
  - MINIMIZE COMPONENT FLOW RATE
- OPERATING CAPABILITY WITH OFF-NOMINAL INLET CONDITIONS
  - PROPELLANT INLET TEMPERATURE
  - PROPELLANT INLET PRESSURE
- EVALUATION OF PROPELLANT CONTROL VALVE ACTUATION
  - HIGH FLOW AREA VALVES REQUIRE HIGH SEAT LOADS
  - RESPONSE REQUIREMENTS IMPOSE HIGH ACTUATOR DYNAMIC LOADS

FIGURE 6-1

LOW PRESSURE ENGINE TECHNOLOGY REVIEW

DESIGN EVALUATION	KEY TECHNOLOGY AREAS
o HIGH PERFORMANCE INSULATION o INSULATION MUST BE REUSABLE o PRESSURIZED EXTERNAL SHELL ALLOWS THE USE OF HPI	o REQUIRES ON ORBIT VENTING AND REPRESSURIZATION o POTENTIAL CONDUCTIVITY AND REFLECTOR EMISSIVITY DEGRADATION WITH REPEATED VENT AND PRESSURIZATION CYCLES, THERMAL CYCLES AND/OR LONG TERM CORROSION o EVALUATION AND DEMONSTRATION REQUIRED
o PRESSURIZATION o AUTOGENOUS HYDROGEN PRESSURIZATION PROVIDES SIGNIFICANT WEIGHT REDUCTION	o CONTROL OF TANK THERMODYNAMICS REQUIRED TO PREVENT EXCESSIVE VENT LOSSES AND/OR INTERACTIONS WITH PROPELLANT ACQUISITION SYSTEM.
o PROPELLANT ACQUISITION o SCREEN LINERS AND/OR REFILLABLE SCREEN TANK ASSEMBLIES REQUIRED FOR EFFICIENT PROPELLANT STORAGE AND USAGE	o LARGE DIAMETER AND AREA REQUIREMENTS EXCEED CURRENT PRODUCTION EXPERIENCE o DEMONSTRATION UNDER 1_G AND SIMULATION OF ZERO G REQUIRED o SCREEN AND LIQUID RESONANT FREQUENCIES MUST BE MAINTAINED OUTSIDE OF FLIGHT VIBRATION ENVIRONMENT

FIGURE 6-2

LIQUID STORAGE TECHNOLOGY REVIEW

DESIGN EVALUATION	KEY TECHNOLOGY AREAS
° FEASIBLE DESIGN APPROACH	° MULTIPLE FLEXIBLE JOINTS: POTENTIAL SEALING PROBLEM IN ATTACHMENT OF HEAT EXCHANGER TUBES TO REPLACEABLE SKIN PANELS
° CONTROL AND STABILITY	° VARIABLE HEAT INPUT AND CONDITIONING DEMANDS: UNCONTROLLED POSITIONING OF FLUID PHASE CHANGE-POSSIBLE CHUGGING INSTABILITY

PASSIVE HEAT EXCHANGER TECHNOLOGY REVIEW

FIGURE 6-3

DESIGN EVALUATION

- o FEASIBLE DESIGN POINT CONDITIONS
- o COMPONENT SIZE PRIMARILY CONTROLLED BY PROPELLANT (COLD SIDE) FLOW RATE AND PRESSURE
- o ACCEPTABLE MATERIALS REQUIREMENTS
- o PRELIMINARY DESIGN DOES NOT ADDRESS POTENTIAL HOT GAS SIDE "ICING"

KEY TECHNOLOGY AREAS

- o TWO-PHASE FLOW REQUIRES FLOW STABILIZATION TO ENSURE PROPER POSITIONING OF PHASE CHANGE
- o TUBE EXTERNAL WALL TEMPERATURE CONTROL TO INHIBIT FREEZING COMBUSTION PRODUCTS (H_2O)
  - PROPELLANT/HOT GAS SEQUENCING TO HEAT EXCHANGER
  - TUBE WALL THERMAL CONDUCTIVITY "TAILORING" FROM LIQUID INLET
- o HIGH HEAT TRANSFER FILM COEFFICIENTS REQUIRED ON HOT SIDE WITH LOW PRESSURE DROP
  - BAFFLING HOT GAS FLOW
  - HOT GAS VELOCITY CONTROL

FIGURE 6-4

LOW PRESSURE ACTIVE HEAT EXCHANGER TECHNOLOGY REVIEW

DESIGN POINT EVALUATION	KEY TECHNOLOGY AREAS
- PROPELLANT MIXING TO ATTAIN PRESCRIBED PROPELLANT TEMPERATURE CURRENTLY USED IN ALRC O_2/H_2 TEST FACILITIES	- LOW PRESSURE PHASE CHANGE MIXERS SUSCEPTIBLE TO LIQUID-CHOKED FLOW AND CHUGGING
- DESIGN CONCEPT IS FEASIBLE	- REQUIRES EVALUATION AND DEMONSTRATION TO ASSESS MIXING AND ABILITY TO OBTAIN HOMOGENOUS TEMPERATURE AND PRESSURE IN SHORT FLOW LENGTHS

FIGURE 6-5

LIQUID-VAPOR PROPELLANT MIXER TECHNOLOGY REVIEW

DESIGN EVALUATION

MECHANICAL REGULATOR

- NO CURRENT FLIGHT WEIGHT REGULATOR EXISTS FOR COMBINATION OF OPERATING REQUIREMENTS
  - SUBCRITICAL PRESSURE RATIO
  - LARGE VOLUMETRIC FLOW RATE
  - LOW PRESSURE DROP
  - CLOSE CONTROL PRESSURE TOLERANCE
- REGULATED PRESSURE RATIO DEMONSTRATED WITH HIGHER PRESSURE LEVEL DESIGNS
- PROPELLANT INLET TEMPERATURE VARIABILITY INFLUENCES CONTROL CAPABILITY
- LOW SUPPLY PRESSURE MINIMIZES ACCURACY OF ACTUATOR INTERNAL FORCE BALANCES

IRIS REGULATOR

- FLOW CONTROL CONCEPT DEMONSTRATED IN SIZE REQUIRED
- REQUIRED RESPONSE FASTER THAN CURRENT CAPABILITY
- CYCLE LIFE CAPABILITY DEPENDENT UPON RESPONSE CRITERIA AND ACTUATOR DESIGN CONCEPT

KEY TECHNOLOGY AREAS

- REQUIRES EVALUATION AND DEMONSTRATION OF SUBCRITICAL FLOW REGULATOR CONTROL DEVICE WITH MAXIMUM FLOW AREA
- ACTUATOR POSITIVE CONTROL DEVICE REQUIRED TO MINIMIZE INACCURACY DUE TO INTERNAL FORCE BALANCES

- DEMONSTRATION OF CONTROL SUBSYSTEM FOR FEED SYSTEM CONTROL CAPABILITY AND DYNAMIC RESPONSE
  - IRIS THROTTLE VALVE
  - VARIABLE FEED SYSTEM GEOMETRY
  - VARIABLE FLOW REQUIREMENTS

FIGURE 6-6

LOW PRESSURE REGULATOR TECHNOLOGY REVIEW

7. REFERENCES

(a) Green, W. M., Patten, T. C., Low Pressure Auxiliary Propulsion Subsystem Definition Subtask B Report: McDonnell Douglas Report MDC E0302, dated 29 January 1971

APPENDIX A: SPACE SHUTTLE DESCRIPTION AND REQUIREMENTS

A-1. INTRODUCTION

This appendix describes space shuttle configurations and mission requirements used as a basis for Subtask A Auxiliary Propulsion Subsystem (APS) design definition studies. These shuttle configurations and requirements were selected by the NASA to represent a broad range of shuttle design approaches and were therefore incorporated as part of the contract statement of work (SOW). Two baseline space shuttles were considered (Shuttle A and B), each consisting of reusable booster and orbiter stages. Shuttle A orbiter had a low cross-range capability, and was designed to reenter at a high angle of attack to minimize vehicle heating rates and temperatures. Shuttle B orbiter had a high cross-range capability and was designed for good hypersonic flight performance. Presented in the following paragraphs are overall vehicle/mission requirements, subsystem and component design criteria, and baseline shuttle configurations and characteristics. Two deviations from the requirements of this appendix were made with the consent of the NASA technical director:

(1) Engine locations were modified to minimize control axis cross-coupling in the event one or two engines failed closed.

(2) The same thrust level was employed for all engines to minimize development requirements.

A-2. MISSION/SHUTTLE REQUIREMENTS

Mission characteristics applicable to both shuttles are shown in Figure A-1. The reference mission used for the Subtask A APS study was the logistics resupply of a space station or space base. The reference orbit was a 270 nautical mile circular orbit with a 55 degree inclination. Mission timelines for the space station/base logistics mission are presented in Figures A-2 and A-3 for the orbiters, and Figures A-4 and A-5 for the boosters. These timelines provide data on the range of ΔV requirements and the vehicle attitude limits for nominal missions. Whereas only a single mission timeline was specified for the boosters, three distinct timelines were specified for the orbiters, corresponding to the number of +X axis maneuvers to be performed by the APS. The three orbiter APS maneuver levels considered were:

(1) APS satisfies all maneuver requirements except +X maneuvers greater than 10 ft/sec.

(2) APS satisfies all maneuver requirements except +X maneuvers greater than 50 ft/sec.

(3) APS satisfies all maneuver requirements.

The first two options predicate use of a separate orbit maneuver subsystem (OMS) to provide the additional required maneuver ΔV. Limit cycle ΔV was determined by applying the minimum impulse bit/thrust correlation of Figure 3-8 of this report in conjunction with the control deadband requirements of Figures A-2 and A-4. Additional ΔV requirements for attitude maneuvers not specified in the mission timelines, were 10 ft/sec for the orbiters and 4 ft/sec for the boosters. Maneuver acceleration requirements for both the orbiters and boosters are tabulated in Figures A-6 and A-7, respectively.

Orbital Characteristics \ Missions	Space Station Base Logistics Support	Placement and Retrieval of Satellites	Delivery of Propulsive Stages & Payload	Delivery of Propellants	Satellite Service & Maintenance	Short Duration Orb. Mission
Altitude (n. mi.)	200 - 300	100 - 800	100 - 200	200 - 300	100 - 800	100- 300
Inclination (deg)	28.5 - 90	28.5 - Sun Syn.	28.5 - 55	28.5 - 55	28.5 - Sun Syn.	28.5 - 90
On-Orbit ΔV (1000ft)	1 - 2	1 - 5	1 - 1.5	1 - 2	1 - 5	1 - 2
On-Orbit Stay Time (days)	7	7	7	7	7 - 15	7 - 30
Crew	2	2	2	2	2	2
Passengers (min)	Rotate 30 men/qtr	2	2	2	4	12
Discretionary Payload						
Weight (1000 lbs)	*70/qtr	------	------	------	------	------
Volume (1000 ft^3)		5 - 10	10	10	5 - 10	4 - 6
Critical Dimen. Dia. (ft)	10 - 15	15	15	15	15	15
Return Discretionary Payload						
Weight (1000 lbs)	*45/qtr	------	------	------	------	------
Volume (1000 ft^3)	—	5 - 10	—	8 - 11	5 - 10	4 - 6

Include Passengers

FIGURE A-1

MISSION CHARACTERISTICS

EVENT COMPLETION TIME*		EVENT	PROPULSION REQ'T DESCRIPTION
1	-165 sec.	Staging	Separation of booster and orbiter (No APS requirement)
2.	0	Insertion into nominal 50 × 100 N.M. orbit	Damping of main engine cutoff transients.
3.	-	Manual attitude hold	±45° deadband
4	-	Orient to burn attitude and establish Fine Attitude Hold	0.5° deadband
5.	15-46 min.	Circularization at 100 N.M.	100 - 130 fps ΔV
6.	-	Manual attitude hold	45° deadband
7.	-	Orient to burn attitude and establish Fine Attitude Hold	0.5° deadband
8	59-524	Dispersion on 100 N.M. circularization burn	0 - 32 fps ΔV

* Time is referenced to Event 2 in minutes unless otherwise stated. Both minimum and maximum cumulative times are shown.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE - ORBITER

FIGURE A-2

	EVENT COMPLETION TIME	EVENT	PROPULSION REQ'T DESCRIPTION
9.	-	Drifting flight	No deadband
10.	-	Orient to burn attitude and establish Fine Attitude Hold	±0.5° deadband
11.	74-1005	Plane change burn	0 - 200 fps Δ V
12.	-	Manual attitude hold	±45° deadband
13.	-	Orient to burn attitude and establish Fine Attitude Hold	± 0.5° deadband
14.	84-1165	Dispersion on plane change burn	0 - 50 fps Δ V
15.	-	Drifting flight	No deadband
16.	-	Orient to burn attitude and establish Fine Attitude Hold	±0.5° deadband
17.	94-1326	Transfer to 100 × Phasing Orbit Altitude	279 - 290 fps Δ V
18.	-	Manual attitude hold	±45° deadband
19.	-	Orient to burn attitude and establish Fine Attitude Hold	±0.5° deadband

Mission Timeline

FIGURE A-2

	EVENT COMPLETION TIME	EVENT	PROPULSION REQ'T DESCRIPTION
20.	116-1349	Dispersions on transfer burn	0 - 72 fps ΔV
21.	-	Manual attitude hold	$\pm 45°$ deadband
22.	-	Orient to burn attitude and establish Fine Attitude Hold	$\pm 0.5°$ deadband
23.	138-1372	Circularization at Phasing Orbit Altitude	253 - 297 fps ΔV
24.	-	Manual attitude hold	$\pm 45°$ deadband
25.	-	Orient to burn attitude and establish Fine Attitude Hold	$\pm 0.5°$ deadband
26.	195-1429	Dispersions on circularization burn	0 - 74 fps ΔV
27.	-	Manual attitude hold	$\pm 45°$ deadband
28.	-	Orient to burn attitude and establish Fine Attitude Hold	$\pm 0.5°$ deadband
29.	253-1487	TPI	30-37 fps ΔV
30.	-	Coarse automatic attitude hold	$\pm 5°$ deadband

Mission Timeline

FIGURE A-2

	EVENT COMPLETION TIME	EVENT	PROPULSION REQ'T DESCRIPTION
31.	—	Orient to burn attitude and establish Fine Attitude Hold	±0.5° deadband
32.	275-1510	MCC-1	0 - 36 fps ΔV
33.	—	Coarse automatic attitude hold	±5° deadband
34.	—	Orient to burn attitude and establish Fine Attitude Hold	±0.5° deadband
35.	286-1522	MCC-2	0 - 19 fps ΔV
36.	—	Coarse automatic attitude hold	±5° deadband
37.	—	Orient to burn attitude and establish Fine Attitude Hold	±0.5° deadband
38.	297-1533	TPF	28 - 30 fps ΔV
39.	—	Stationkeeping and attitude hold	0-10 fps Multiaxis Translation ΔV and 0-10 fps Multiaxis Attitude ΔV (±0.5° deadband)
40.	312-1593	Docking	0-10 fps Multiaxis Translation ΔV and 0-10 fps Multiaxis Attitude ΔV (±0.5° deadband)
41.	7277-8505	On Orbit	(TBD)
42.	7227-8508	Undock	0.5 fps ΔV
43.	—	Fine Attitude Hold	±0.5° deadband

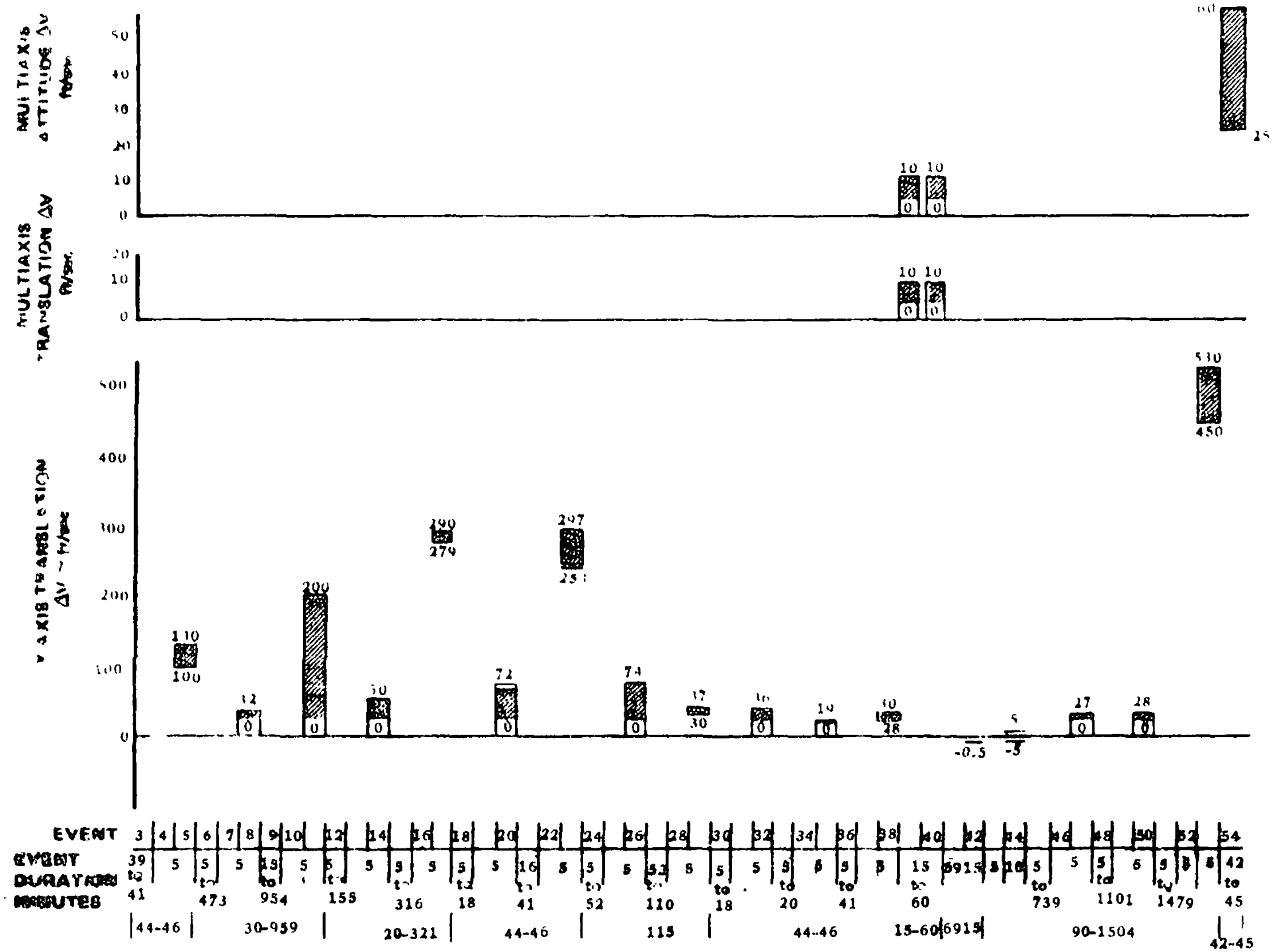
Mission Timeline

FIGURE A-2

EVENT COMPLETION TIME	EVENT	PROPULSION REQ'T DESCRIPTION
44. 7242-8523	Orient to burn attitude and establish Fine Attitude Hold and Separation	±0.5° deadband 10 fps ΔV
45. -	Course attitude hold	±45° deadband
46. -	Orient to burn attitude and establish Fine Attitude Hold	±0.5° deadband
47. 7279-9267	Ground track adjust #1	0-27 fps ΔV
48. -	Manual attitude hold	±45.0° deadband
49. -	Orient to burn attitude and establish Fine Attitude Hold	± 0.5° deadband
50. 7297-9639	Ground track adjust #2	0-28 ΔV
51. -	Manual attitude hold	±45° deadband
52. -	Orient to burn attitude and establish Fine Attitude Hold	±0.5° deadband
53. 7317-10012	De-orbit	450-530 fps ΔV
54. 7359-10057	Entry	25-60 fps ΔV

Mission Timeline

FIGURE [illegible]



SPACE STATION/BASE LOGISTICS MISSION TIMELINE - ORBITER

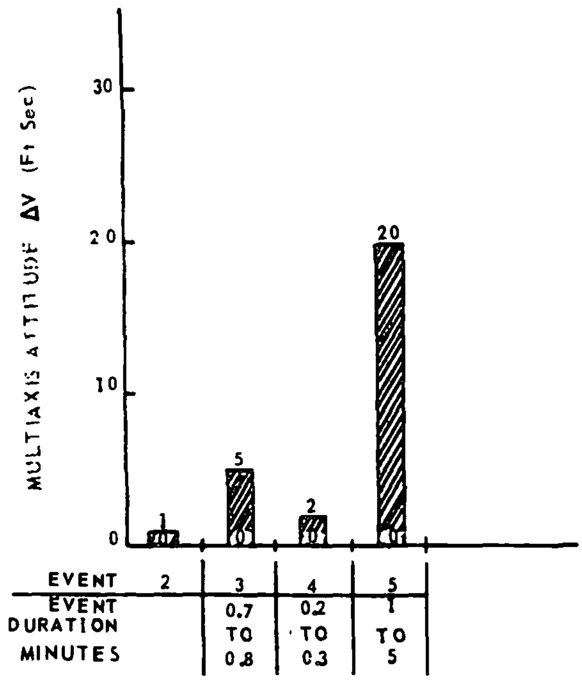
FIGURE A-3

	EVENT COMPLETION TIME*	EVENT	PROPULSION REQUIREMENT DESCRIPTION
1.	0	Staging	Separation of booster and orbiter (No APS requirement)
2.	0+	Post Separation	Damping of main engine cutoff and separation transients.
3.	0.7-0.8	Orientation	Maneuver vehicle to reentry attitude.
4.	0.9-1.1	Attitude hold	±2° deadband
5.	1.9-6.1	Entry	±2° deadband

*Time is referenced to Event 1 in minutes unless otherwise stated. Both minimum and maximum cumulative times are shown.

SPACE STATION/BASE LOGISTICS MISSION TIMELINE – BOOSTER

FIGURE A-4



SPACE STATION/BASE LOGISTICS MISSION TIMELINE - BOOSTER

FIGURE A-5

EVENT[a]		2nd STAGE BOOST ENGINE OUT			1 TO 28 46 TO 53			29 TO 45			54		
		X	Y	Z	X[b]	Y	Z	X	Y	Z	X	Y	Z
TRANSLATION ACCELERATION ft/sec²	MIN		NO REQUIREMENT		0.07	0.07	0.07		SAME AS EVENT 1			NO REQUIREMENT	
	NOM MIN				0.1	0.1	0.1						
	NOM MAX				0.5	0.25	0.25						
	MAX				1.0	1.0	1.0						
		R	P	Y	R	P	Y	R	P	Y	R	P	Y
ANGULAR ACCELERATION deg/sec²	MIN		NO REQUIREMENT		0.3	0.3	0.3		SAME AS EVENT 1		0.3	0.3	0.3
	NOM MIN				0.5	0.5	0.5				1.0	0.5	1.00
	NOM MAX				2.0	2.0	2.0				1.75	1.0	1.75
	MAX				4.0	4.0	4.0				4.0	4.0	4.0
FINE ATTITUDE LIMITS — deg					0.5	0.5	0.5	0.5	0.5	0.5	2.0	2.0	2.0
COARSE ATTITUDE LIMITS — deg					45	45	45	5.0	5.0	5.0			

a. Refer to Table I-2 for event names corresponding to these event numbers.

b. Maximum limits for events 5, 8, 11, 14, 17, 20, 23, 26, and 53 may be increased when the APS is used to perform the major plus X translation.

FIGURE A-6

SPACE STATION/BASE LOGISTICS MISSION — ORBITER MANEUVERING CAPABILITY REQUIREMENTS

EVENT[a]		2			3			4			5		
		X	Y	Z	X	Y	Z	X	Y	Z	X	Y	Z
TRANSLATION ACCELERATION ft/sec²	MIN NOM MIN NOM MAX MAX	NO REQUIREMENT						NO REQUIREMENT					
		R	P	Y	R	P	Y	R	P	Y	R	P	Y
ANGULAR ACCELERATION deg/sec²	MIN NOM MIN NOM MAX MAX	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0	0.3 0.5 1.0 2.0	0.3 1.0 1.75 2.0
ANGULAR RATE deg/sec	MIN NOM MIN NOM MAX MAX				 2 	 2 							
ATTITUDE LIMITS — deg		2	2	2	2	2	2	2	2	2	2	2	2

a. Refer to Table I-3 for Event names corresponding to these numbers.

FIGURE A-7

SPACE STATION/BASE LOGISTICS MISSION — BOOSTER MANEUVERING CAPABILITY REQUIREMENTS

A-3. SUBSYSTEM/COMPONENT DESIGN CRITERIA

The APS is constrained to operate for a minimum of 100 mission cycles over an eight year period without major overhaul or refurbishment. Additionally, the APS must fail operational after failure of any critical component, and after the second failure must provide safe operation for crew survival. Specific criteria applied to APS design are summarized below:

(1) A material ultimate factor of 2.0 was applied to stresses resulting from pressure conditions, separate from other loads.

(2) No propellant reserve was added to that estimated to be consumed in a mission cycle.

(3) Fluid line sizes were selected on the basis of design velocities that resulted in acceptable compromise between the excessive pressure drop produced by small diameter tubing and the weight and cost of large diameter tubing. Gas velocities were limited to Mach 0.3 or less.

(4) Flight components were limited to those required for flight operations, except for components required for on-board checkout and servicing. Component integration, packaging, and simplicity of checkout were considered where advantages in maintainability, serviceability, replaceability, weight and cost could be realized.

(5) Provisions for ease of inspection, servicing and maintenance were incorporated in the subsystems and component design to meet the short turn-around and launch preparation requirement.

A-4. SHUTTLE CHARACTERISTICS

A-4.1 Characteristics of Shuttle A - Shuttle A has two reusable stages with fixed low-sweep wings and similar aerodynamic shapes. The vehicle is configured for vertical takeoff with the orbiter mounted forward on the booster and with the booster and orbiter engines operated sequentially. Total liftoff weight is approximately 3,384,000 pounds with a payload capability of 49,000 pounds up and 25,000 pounds down. Booster/orbiter separation at staging is accomplished by firing a pyrotechnic thruster having a 0.5 foot stroke. The separation rates are controlled by the booster and orbiter auxiliary propulsion subsystem with the booster auxiliary propulsion subsystem also assisting in translational separation. The auxiliary propulsion subsystems on both the booster and orbiter utilize engines burning gaseous hydrogen and oxygen.

Payload is contained within a compartment 15 feet in diameter by 60 feet long. The compartment is exposed by opening clamshell doors on the upper surface of the orbiter. The payload module then pivots forward on a hinge for docking to the space station/base.

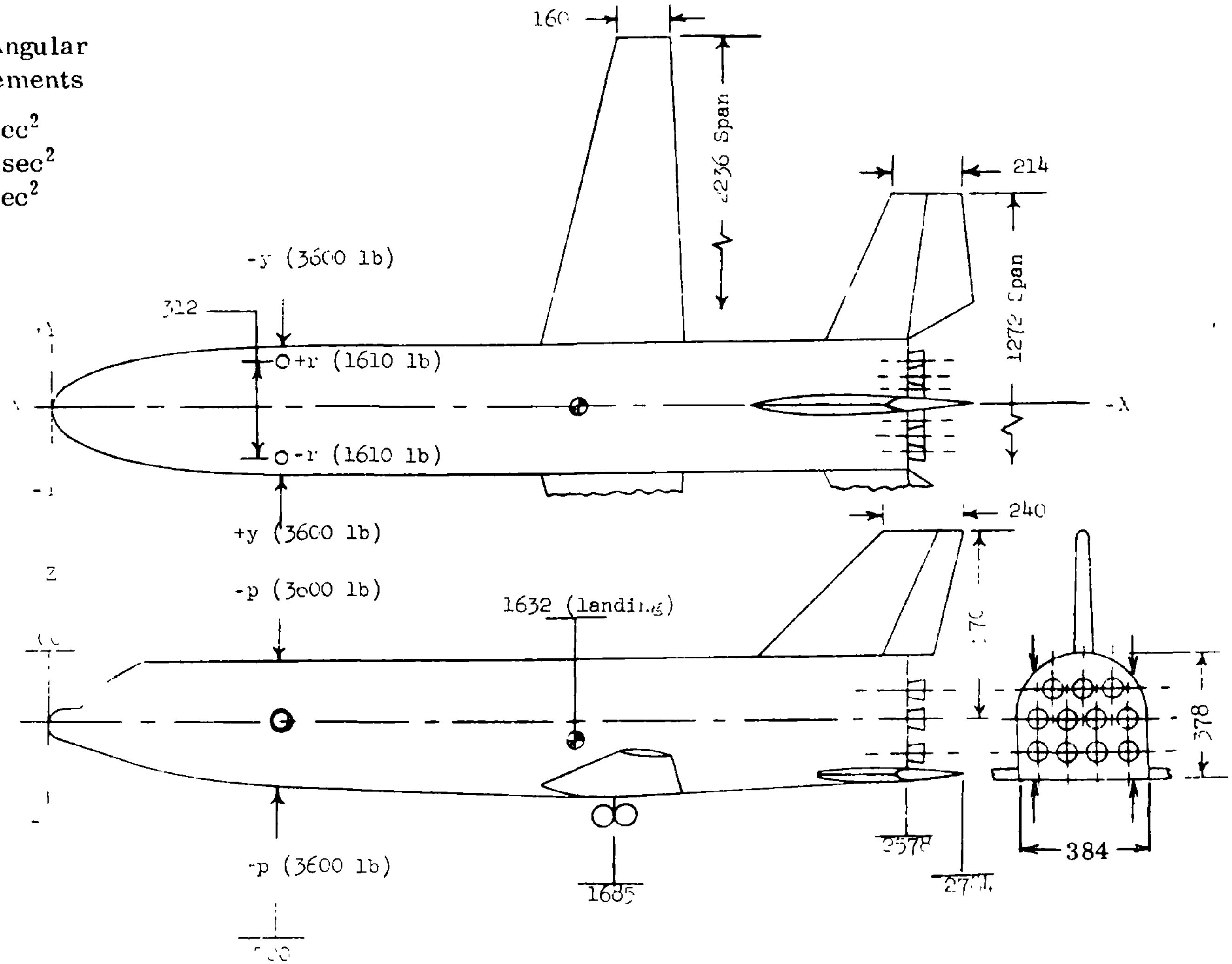
To limit shuttle heating rates and temperatures on reentry, a high angle of attack (60 deg) with high lift coefficient and low lift loading is used for reentering. Both stages must cruise back to a horizontal land landing after reentry. Turbojet engines are used to provide the cruise thrust. Both stages are configured to provide a high subsonic lift to drag ratio for conventional landing attitudes and low landing speeds. Takeoff and self-flight-ferry will be used for earth transport.

Detailed configurational and design data for the booster stage are presented in Figures A-8 through A-13. Similar data for the orbiter stage are provided in Figures A-14 through A-20.

A-4.2 Characteristics of Shuttle B - Shuttle B is a two-stage, fully-reusable, fixed-geometry configuration which is launched vertically and lands horizontally. The two stages are fired sequentially to inject the payload into orbit. The total lift-off weight is approximately 3,500,000 lb. The first stage is a two-bodied configuration whose aerodynamic characteristics provide a high subsonic L/D for fly back, and provides a low attitude at landing. The stage is capable of takeoff and self-flight without modification.

Engine Thrust Shown
Based on Minimum Angular
Acceleration Requirements

Roll: 0.3 deg/sec^2
Pitch: 0.3 deg/sec^2
Yaw: 0.3 deg/sec^2

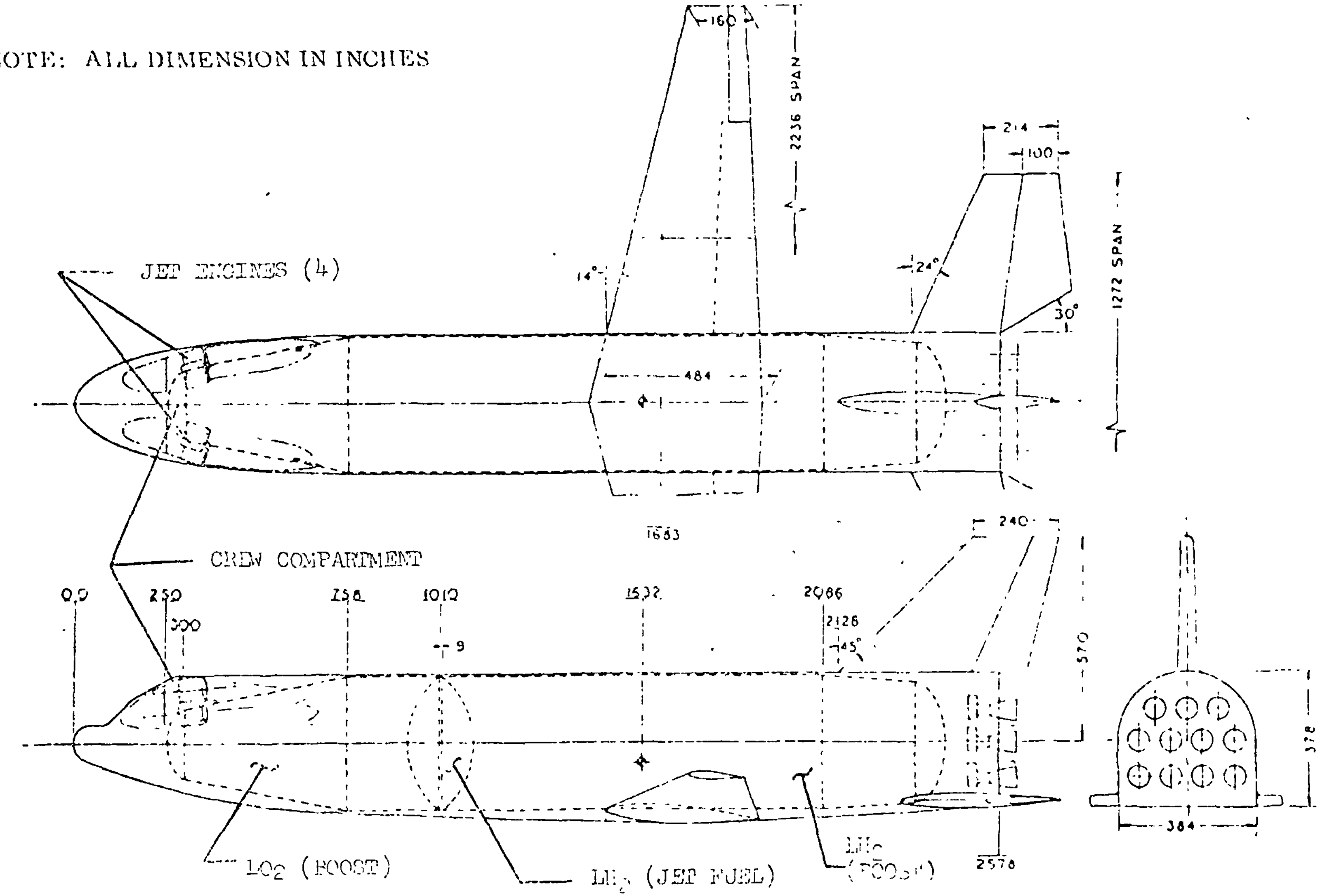


NOTE: ALL DIMENSIONS IN INCHES

BOOSTER EXTERNAL PROFILE — VEHICLE A

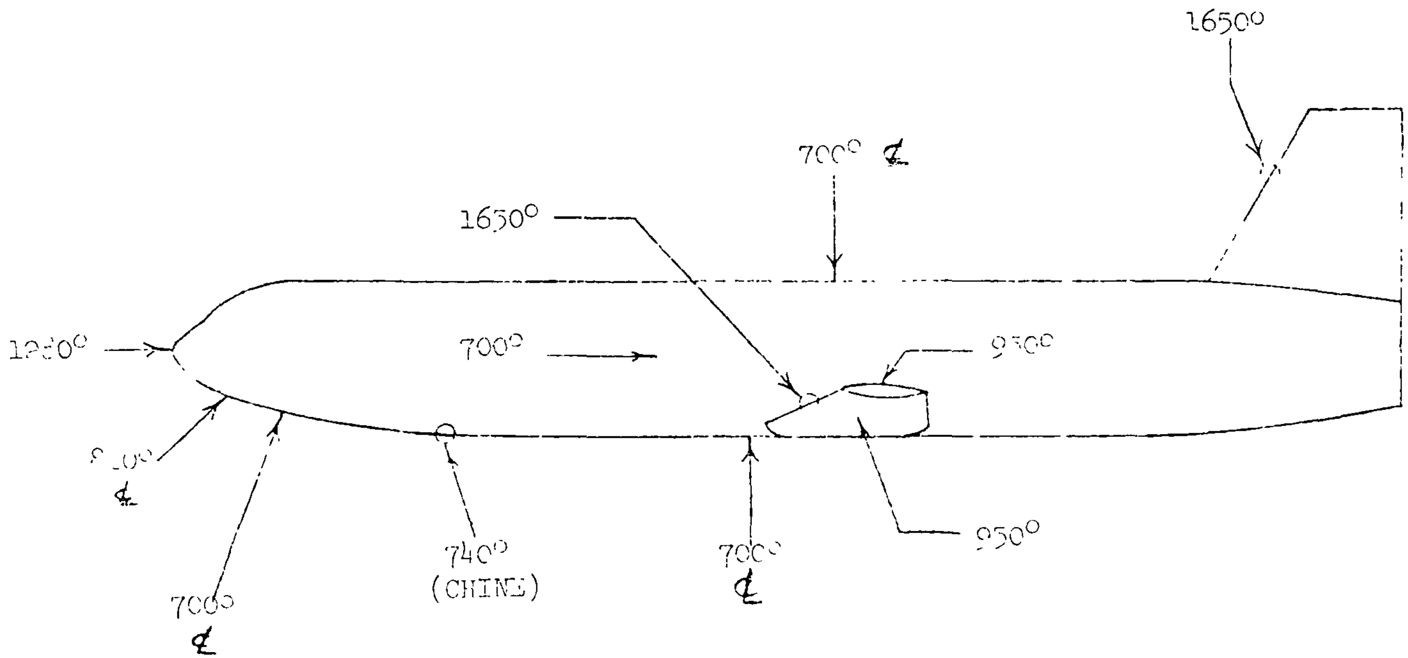
FIGURE A-8

NOTE: ALL DIMENSION IN INCHES



BOOSTER INTERNAL PROFILE — VEHICLE A

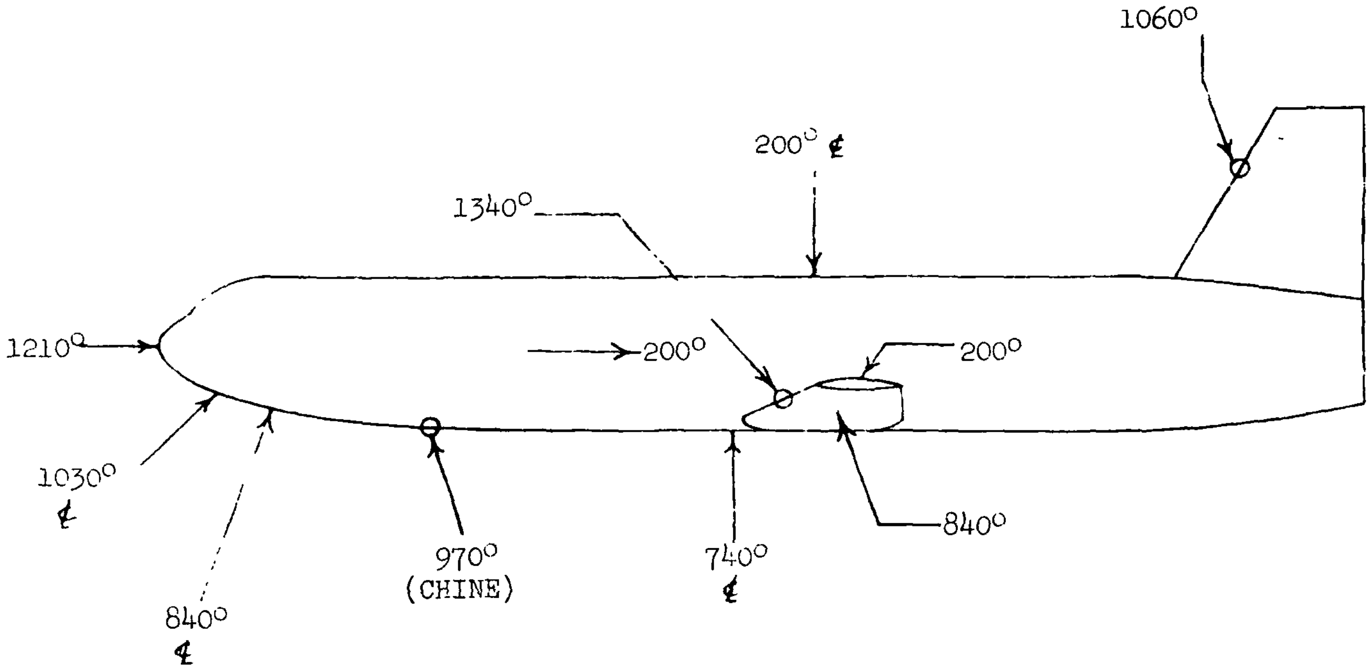
FIGURE A-9



NOTE: ALL TEMPERATURES IN °F

MAXIMUM SURFACE TEMPERATURES DURING ASCENT — BOOSTER — VEHICLE A

FIGURE A-10



NOTE: ALL TEMPERATURES IN °F

MAXIMUM SURFACE TEMPERATURES DURING ENTRY — BOOSTER — VEHICLE A

FIGURE A-11

		STAGING (BOOSTER BURN OUT)	INITIATION OF JET POWERED FLIGHT
WEIGHT — LB		458 600	433 800
CENTER OF GRAVITY LOCATION — INCHES	X	1571.0	1605.9
	Y	0	0
	Z	-45.0	-47.5
MOMENT OF INERTIA — SLUG-FT^2	Ixx	4.060×10^6	3.990×10^6
	Iyy	53.100×10^6	51.700×10^6
	Izz	53.100×10^6	51.700×10^6

MASS CHARACTERISTICS, BOOSTER STAGE — VEHICLE A

	LIQUID HYDROGEN TANK, BOOST	LIQUID OXYGEN TANK, BOOST	LIQUID HYDROGEN TANK, JET FUEL
VOLUME — FT^3	80 652	29 925	6620
PRESSURIZATION BAND — PSIA	40-43	30-33	40-43
VENT PRESSURE BAND — PSIA	43-45	33-35	43-45

PROPELLANT TANKAGE CHARACTERISTICS
BOOSTER STAGE — VEHICLE A

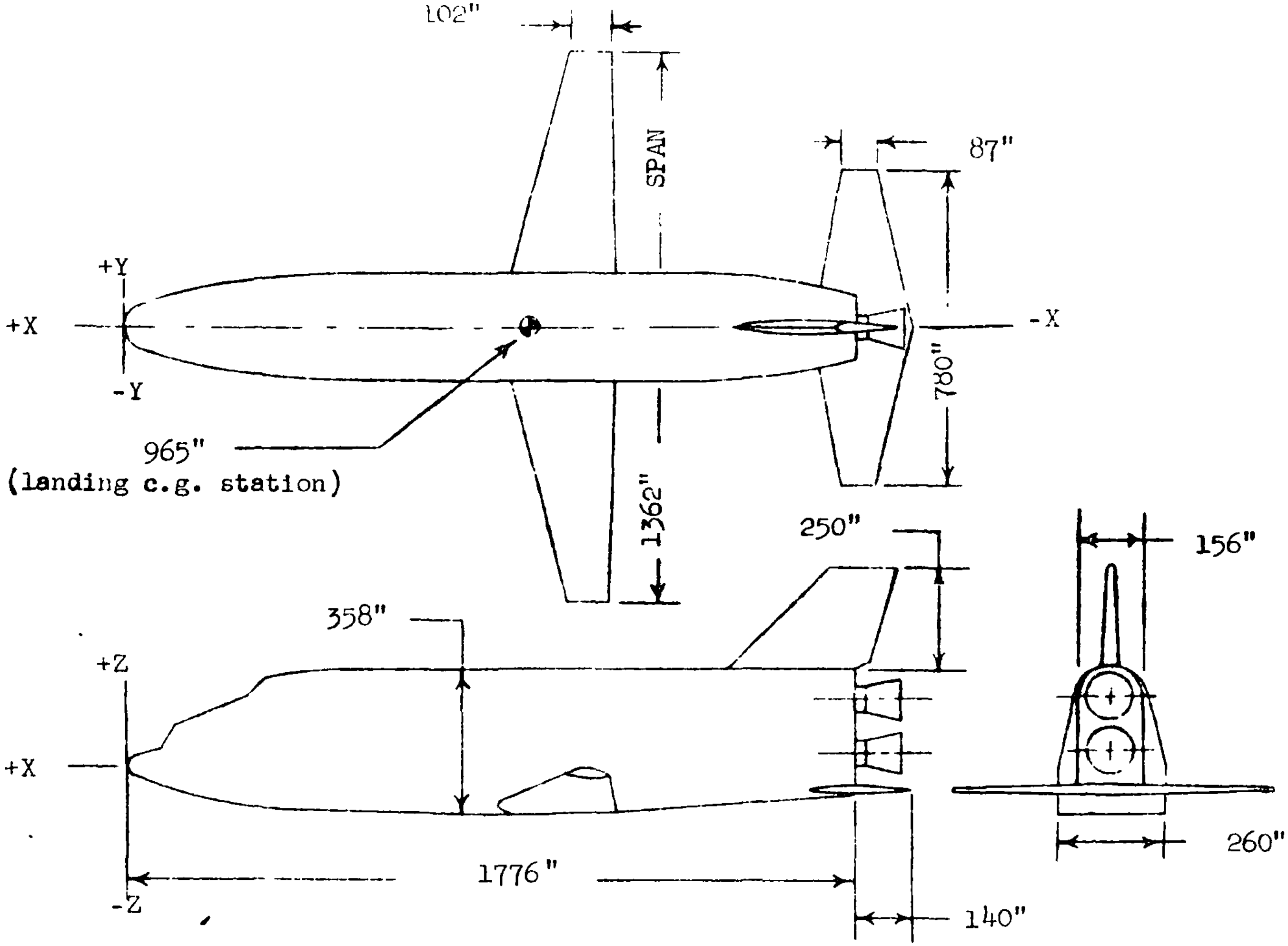
FIGURE A-12

	LIQUID HYDROGEN TANK, BOOST	LIQUID OXYGEN TANK, BOOST
MASS AT BURN OUT – LB	7262	27 202
MASS PERCENT LIQUID – %	52.5	54.5
TEMPERATURE OF VAPOR – °R	200	300
TOTAL PRESSURE – PSIA	45	35
HEAT LEAK INTO PROPELLANT – $BTU/HR/FT^2$	7-36	4-TBD
MASS OF HELIUM PRESSURANT – LB	0[a]	0[a]

a. Autogeneous Pressurization System Assumed

RESIDUAL PROPELLANTS
BOOSTER STAGE – VEHICLE A

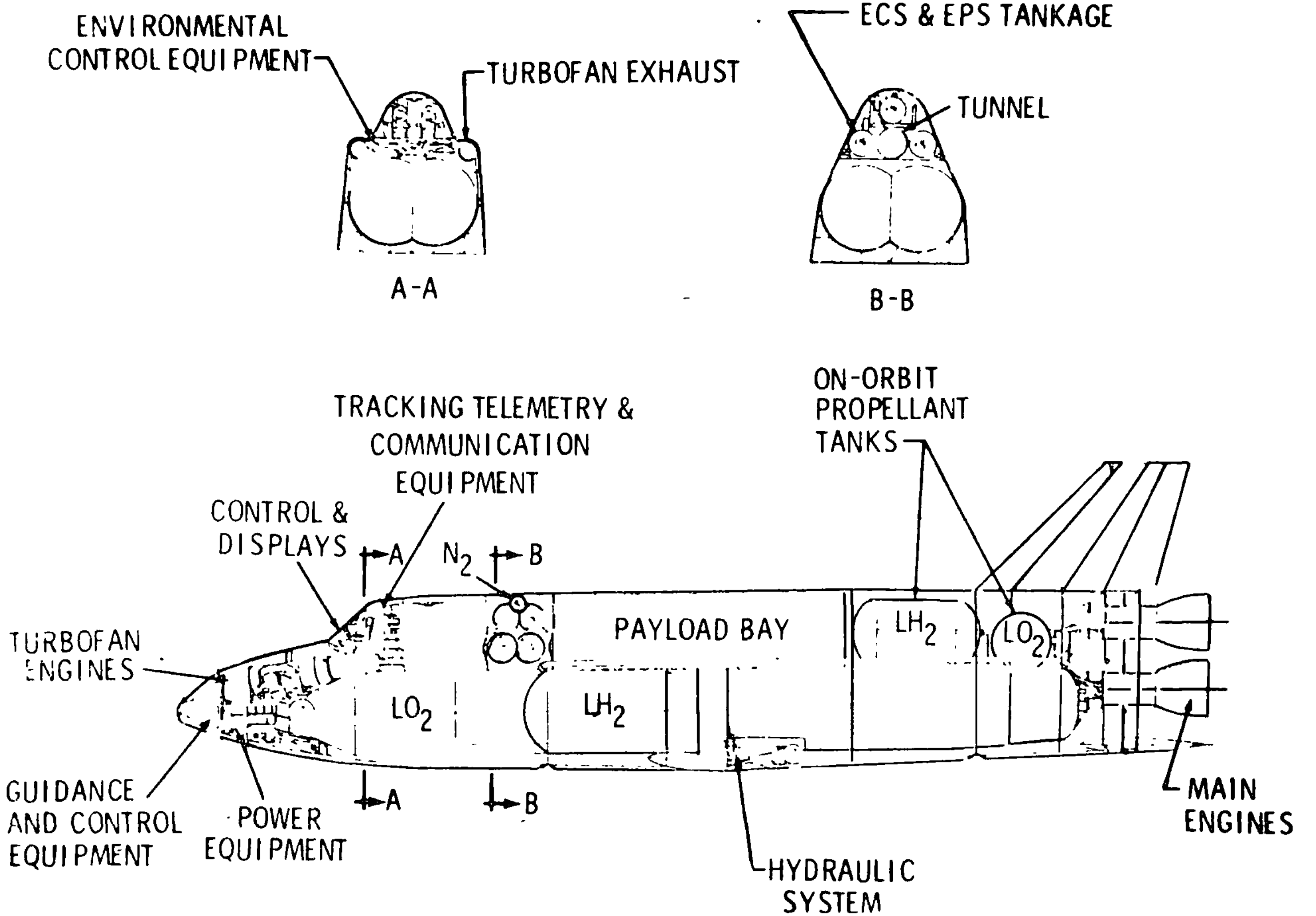
FIGURE A-13



NOTE: ALL DIMENSIONS IN INCHES

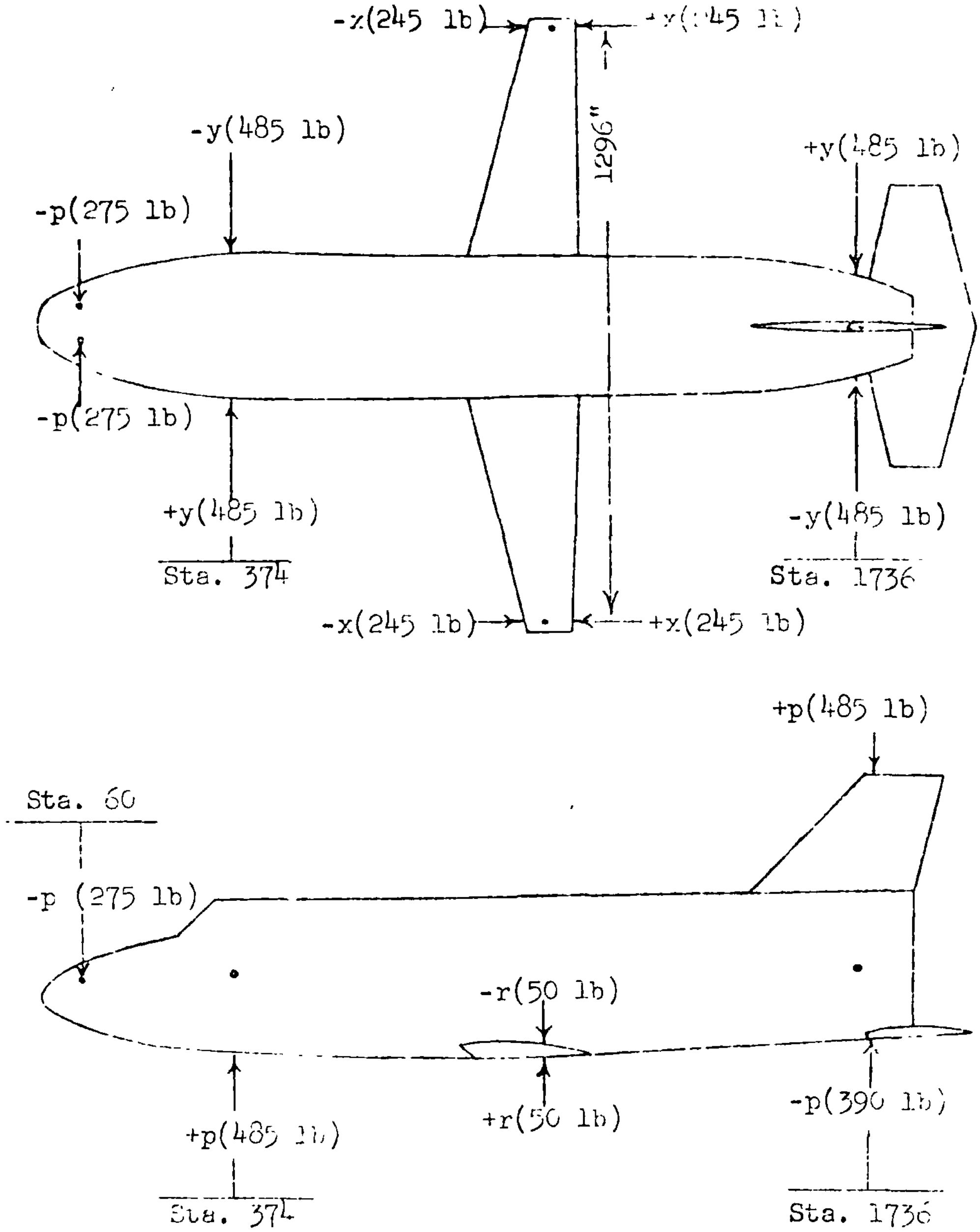
ORBITER EXTERNAL PROFILE — VEHICLE A

FIGURE A-14



ORBITER INTERNAL PROFILE — VEHICLE A

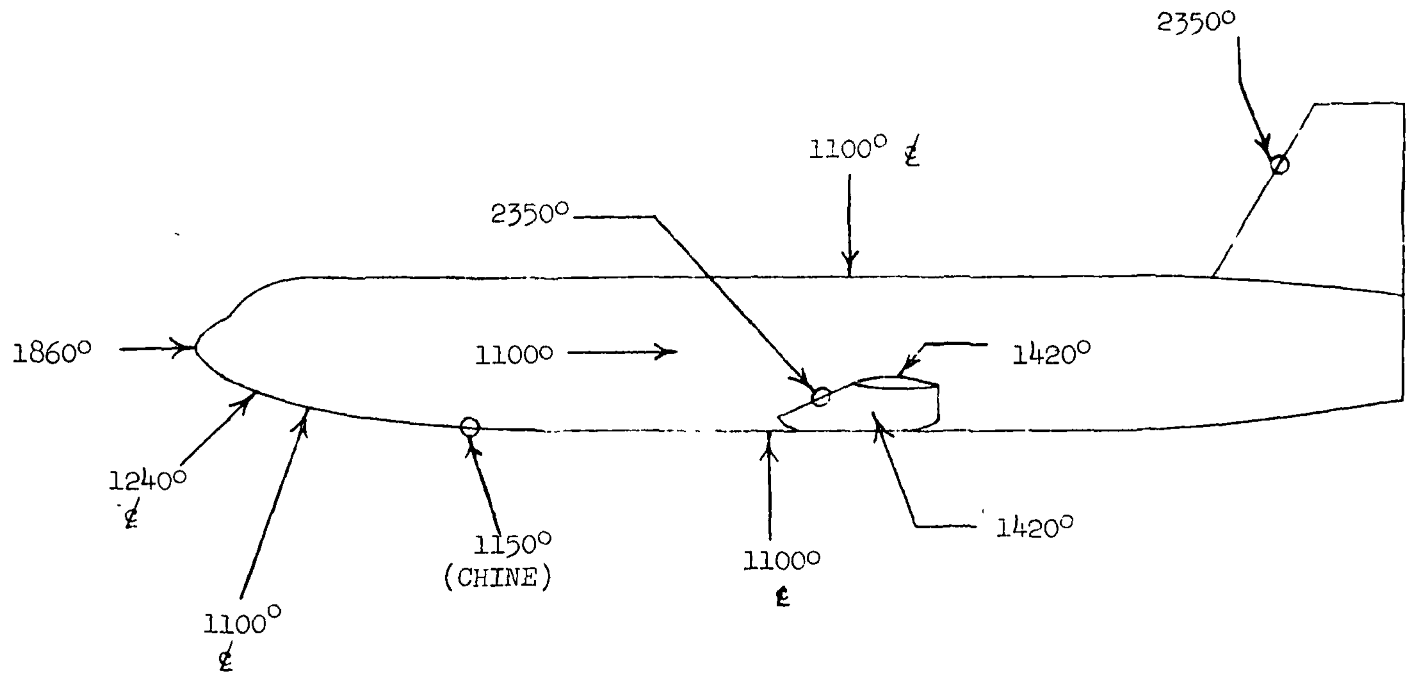
FIGURE A-15



NOTES

1. The forward negative pitch engines are canted outboard 45 deg to limit plume impingement on the deployed payload module
2. The side firing 485 lb thrusters are located at station 75 on the positive Z-axis
3. The forward and reverse firing 245 lb thrusters are located at station 0 on the +Z-axis
4. Engine thrust shown is that required to provide minimum control requirements
   Roll 0.3 deg/sec^2, Pitch 0.3 deg/sec^2 Yaw 0.3 deg/sec^2
   Translation 0.07 ft/sec^2

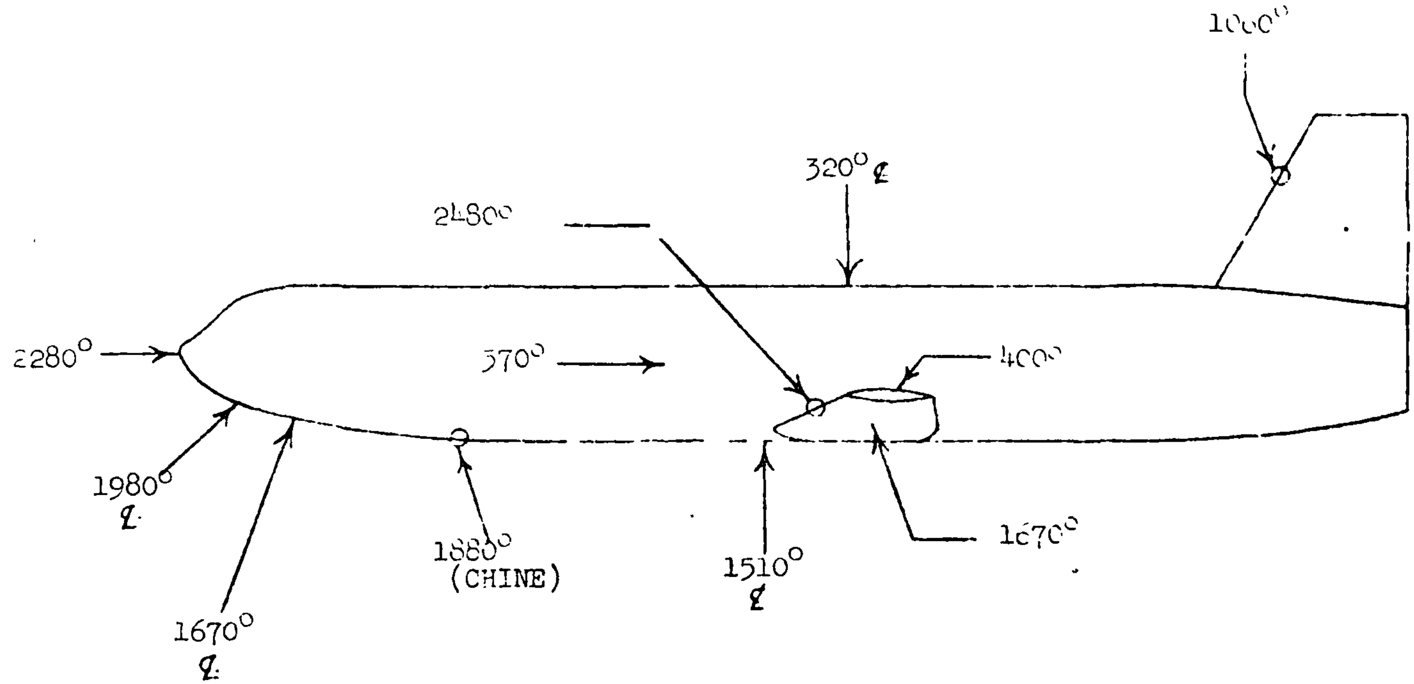
ORBITER APS ENGINE LOCATIONS **FIGURE A-16**



NOTE: ALL TEMPERATURES IN °F.

FIGURE A-17

MAXIMUM SURFACE TEMPERATURES DURING ASCENT — ORBITER — VEHICLE A



NOTE: ALL TEMPERATURES IN °F.

MAXIMUM SURFACE TEMPERATURE DURING ENTRY – ORBITER – VEHICLE A

FIGURE A-18

		SEPARATION	INJECTION (45×100 n. mi.)	INITIATION OF JET POWERED FLIGHT
WEIGHT — LB		624 900	224 900	164 000
CENTER OF GRAVITY LOCATION — INCHES	X	750.2	1040.1	971.3
	Y	0	0	0
	Z^a	29.6	82.9	61.9
MOMENT OF INERTIA — SLUG-FT^2	Ixx	1.427×10^6	1.040×10^6	0.92×10^6
	Iyy	23.565×10^6	10.476×10^6	7.792×10^6
	Izz	23.373×10^6	10.497×10^6	7.901×10^6

a. Z-axis center of gravity measured positive above main propellant tank center line.

MASS CHARACTERISTICS — ORBITER STAGE — VEHICLE A

	LIQUID HYDROGEN TANK, BOOST	LIQUID OXYGEN TANK, BOOST	LIQUID HYDROGEN TANK, ON-ORBIT	LIQUID OXYGEN TANK, ON-ORBIT
VOLUME — FT^3	13 964	5204	1204	271
PRESSURIZATION BAND PSIA	40-43	30-33	40-43	30-33
VENT PRESSURE BAND PSIA	43-45	33-35	43-45	33-35

PROPELLANT TANKAGE CHARACTERISTICS — ORBITER STAGE — VEHICLE A

FIGURE A-19

	LIQUID HYDROGEN TANK, BOOST	LIQUID OXYGEN TANK, BOOST	LIQUID HYDROGEN TANK, ON-ORBIT	LIQUID OXYGEN TANK, ON-ORBIT
MASS AT ORBIT INJECTION – LB	1105	3987		
MASS PERCENT LIQUID – %	46.2	46.0		
TEMPERATURE OF VAPOR – °R	200	300		
TOTAL PRESSURE PSIA	45	35		
HEAT LEAK INTO PROPELLANT BTU/HR/FT^2	7-36	4-TBD	0.5	0.5
MASS OF HELIUM PRESSURANT – LB	0[a]	0[a]	0[a]	0[a]

a. Autogeneous Pressurization System Assumed

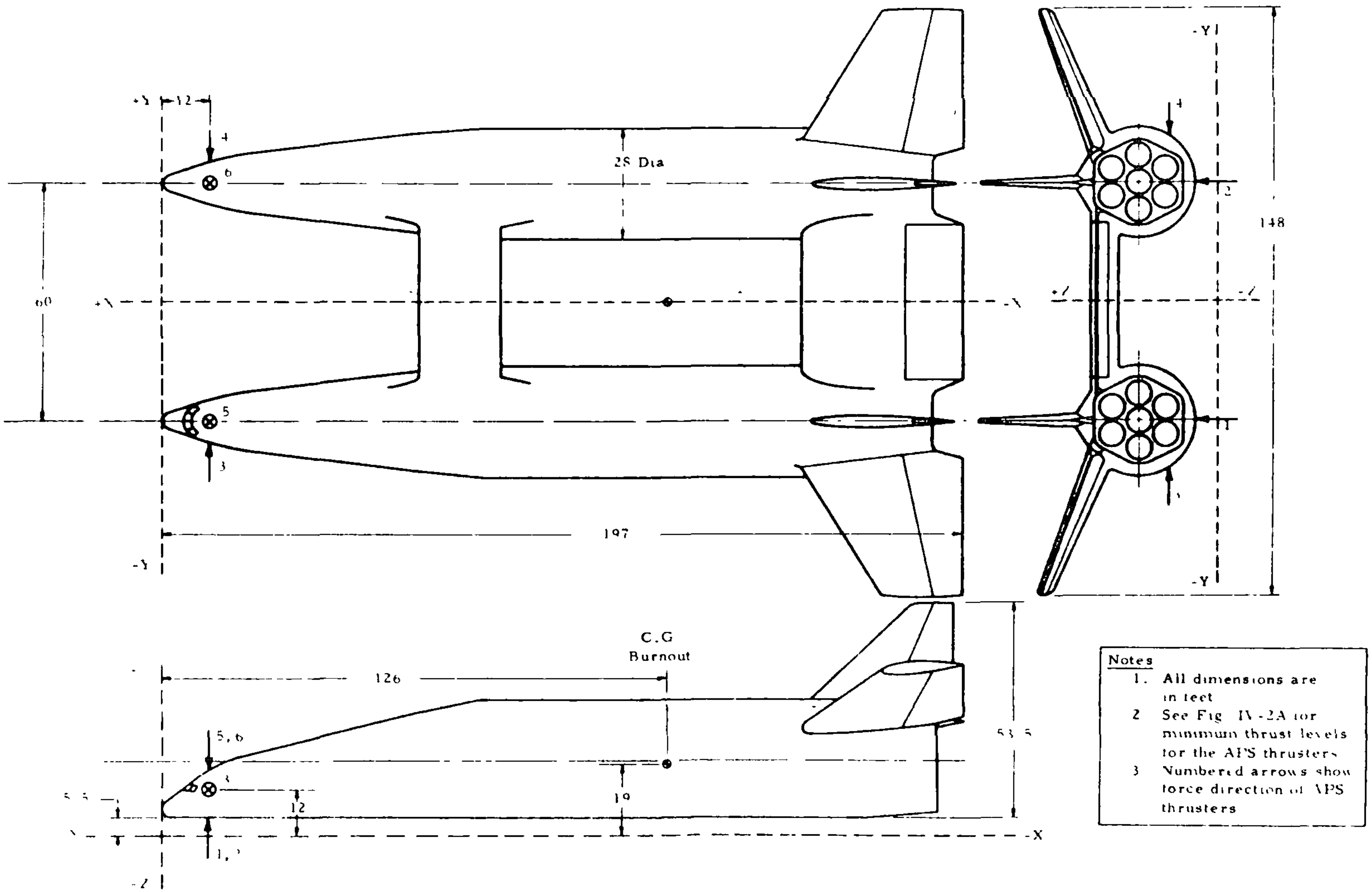
FIGURE A-20

RESIDUAL PROPELLANTS – ORBITER STAGE – VEHICLE A

The second stage is characterized by fixed wing geometry providing good landing characteristics and good hypersonic characteristics for cross range capability.

The two stages are separated during boost phase via translation engines mounted on the booster stage. Orbiter stage ignition is initiated at separation. The booster APS separation engines provide 300,000 lb-sec impulse such that after approximately two seconds from initiation of separation the physical separation of the two stages is sufficient to prevent plume impingement on the booster.

Detailed configurational and design data for the booster stage are presented in Figures A-21 through A-25. Similar data for the orbiter stage are provided in Figures A-26 through A-31.



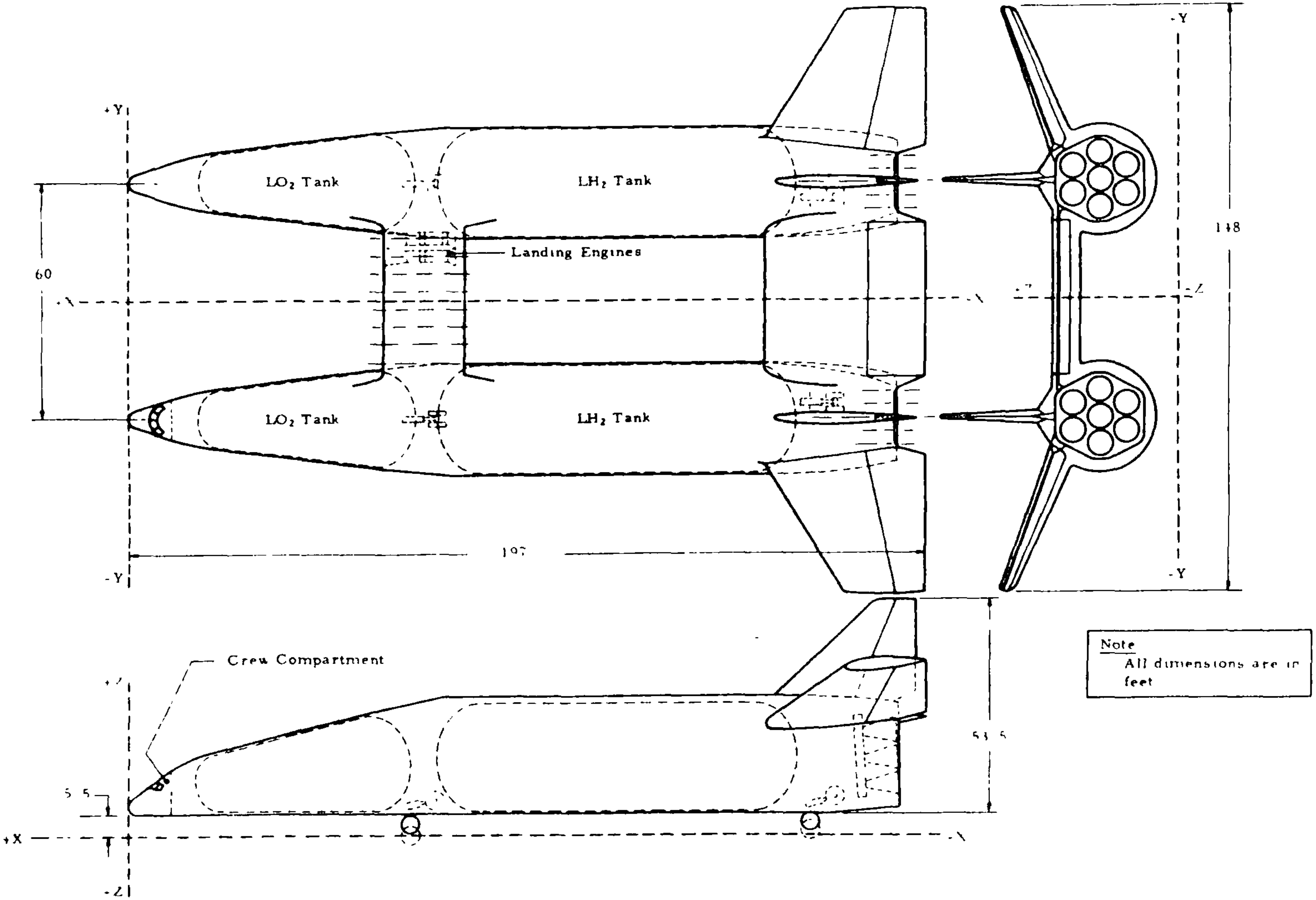
VEHICLE B BOOSTER STAGE — EXTERNAL PROFILE

FIGURE A-21

THRUSTER Fuselage Left	Right	PURPOSE	MINIMUM THRUST LEVEL – LB
1		+ Pitch, + Roll	1785
	2	+ Pitch, - Roll	1785
3		+ Yaw	2344
	4	- Yaw	2344
5		- Pitch, - Roll	1785
	6	- Pitch, + Roll	1785

APS MINIMUM THRUST LEVELS – VEHICLE B – BOOSTER

FIGURE A-22



VEHICLE B BOOSTER STAGE — INTERNAL PROFILE

FIGURE A-23

	Staging (Booster B/O)	Initiation of Reentry	Initiation of Jet Powered Flight
Weight, lbs	497 213	496 200	495 500
Center of Gravity			
X ft	126	126	126
Y ft	0	0	0
Z ft	19	19	19
Moment of Inertia			
Ix-x slug-ft^2	13.329×10^6	13.329×10^6	13.329×10^6
Iy-y slug-ft^2	34.470×10^6	34.470×10^6	34.470×10^6
Iz-z slug-ft^2	45.274×10^6	45.274×10^6	45.274×10^6

MASS CHARACTERISTICS VEHICLE B — BOOSTER

	LOX	LH_2
Volume, ft^3	30 350	95 300
Pressurization Band, psia	20-25	35-40
Vent Pressure Band, psia	26-31	41-46

PROPELLANT TANKAGE VEHICLE B — BOOSTER

		LOX	LH_2
Mass	lbs	20 000	3100
Liquid by Mass	—	65%	19%
Temperature of Vapor	^oR	300	300
Total Pressure	psia	26	41
Heat Leak into Propellant	Btu/hr/ft^2	7-36	4-TBD
Helium Pressurant	lbs	9	190

RESIDUAL PROPELLANTS UPSTREAM OF MAIN ENGINE VALVES AT BURNOUT — VEHICLE B — BOOSTER

FIGURE A-24

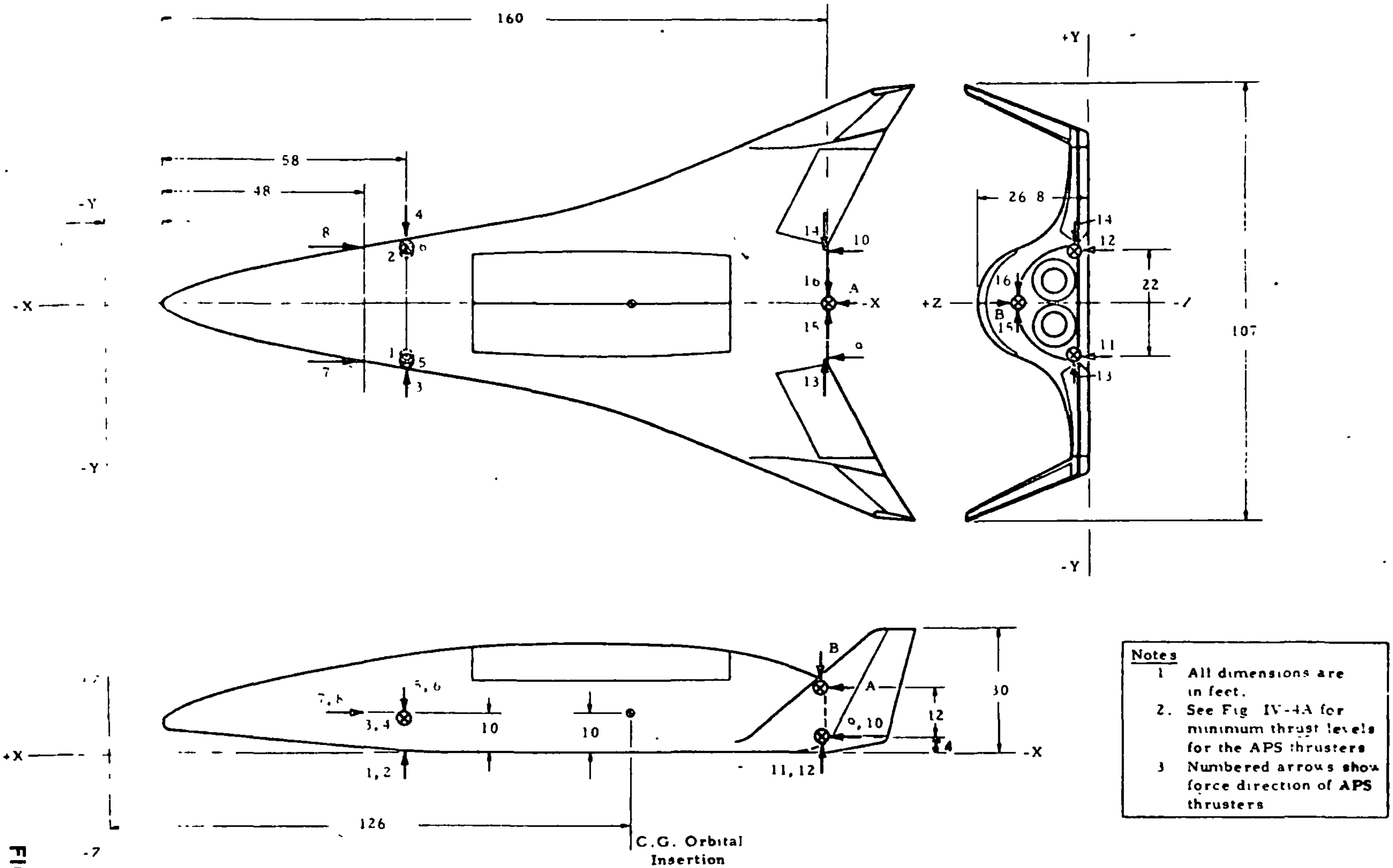
ENTRY			
Station, ft	Position, deg	Q, Btu/ft²/sec	T_w, °R
25	0	1.9	1450
25	180	7.5 13.0	2000 540
25	90/270	1.1 1.9	2000 540
70	180	7.0 13.3	2000 540
184	180	5	1200

Station — Distance along X-axis, ft

Position — Angular position looking in the +X direction, ref: Zero degrees at top

EXTERNAL SURFACE HEATING ENVIRONMENT
VEHICLE B — BOOSTER

FIGURE A-25



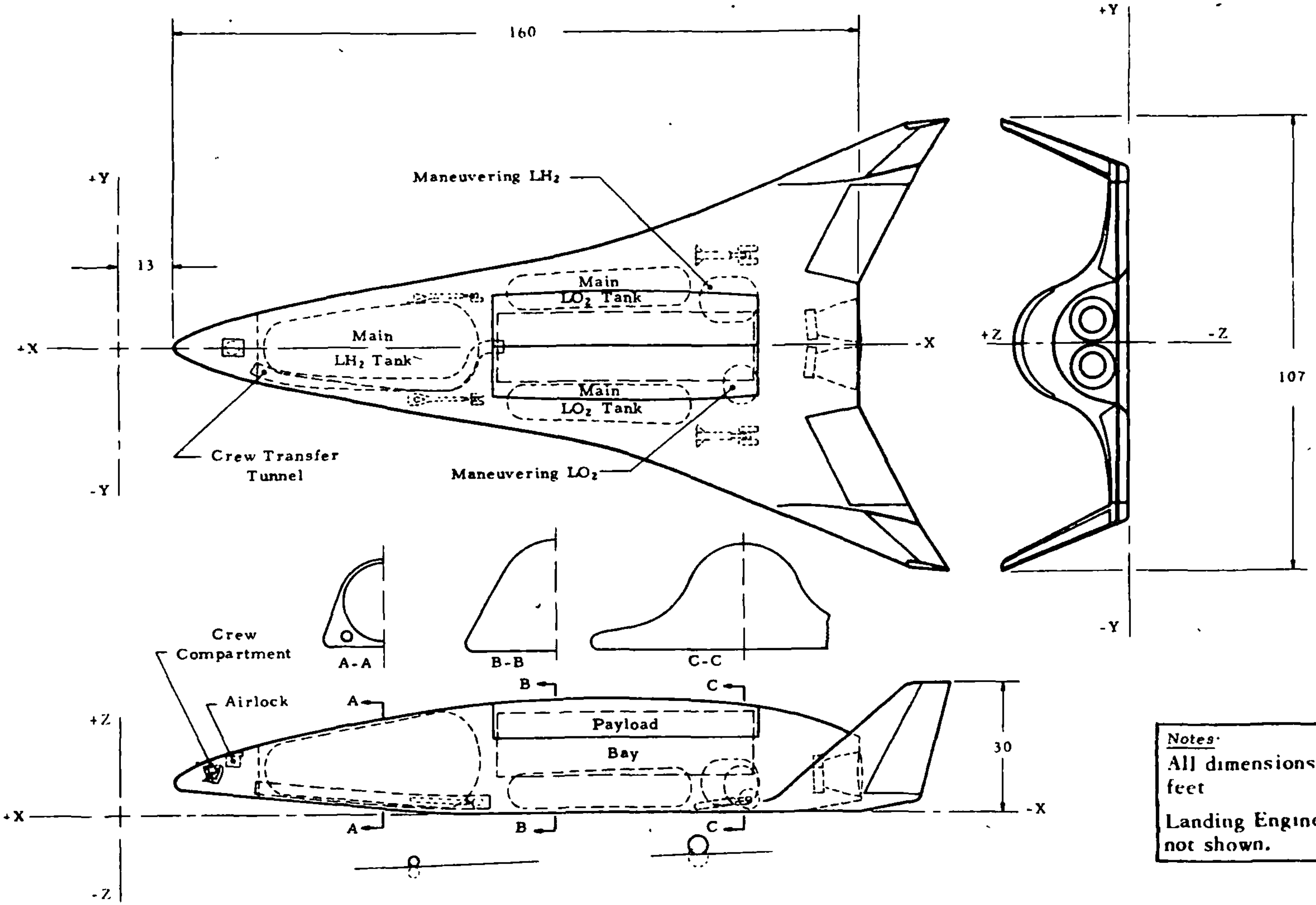
VEHICLE B ORBITER STAGE — EXTERNAL PROFILE

FIGURE A-26

THRUSTER	PURPOSE	MINIMUM THRUST LEVEL – LB
1	+ Pitch, + Z	308
2	+ Pitch, + Z	308
3	+ Yaw, + Y	685
4	- Yaw, - Y	685
5	- Pitch, - Z	308
6	- Pitch, - Z	308
7	- X	279
8	- X	279
9	+ X	186
10	+ X	186
11	- Pitch, + Z	308
12	- Pitch, + Z	308
13	- Yaw, + Y, - Roll	993
14	+ Yaw, - Y, + Roll	993
15	+ Roll	993
16	- Roll	993
A	+ X	186
B	+ Pitch, - Z	618

APS MINIMUM THRUST LEVELS – VEHICLE B – ORBITER

FIGURE A-27



Notes:
All dimensions are in feet
Landing Engines not shown.

VEHICLE B ORBITER STAGE — INTERNAL PROFILE

FIGURE A-28

	Injection 45 × 100 n. mi.	Initiation of Reentry	Initiation of Jet Powered Flight
Weight, lbs	256 790	241 902	231 902
Center of Gravity			
X ft	126	125	124
Y ft	0	0	0
Z ft	10. 0	10. 1	10. 3
Moment of Inertia			
Ix-x slug ft^2	2.28×10^6	2.28×10^6	2.28×10^6
Iy-y slug ft^2	12.078×10^6	12.078×10^6	12.078×10^6
Iz-z slug ft^2	13.357×10^6	13.357×10^6	13.357×10^6

MASS CHARACTERISTICS VEHICLE B — ORBITER

	Main Tanks		**Maneuvering Tanks**	
	LOX	LH_2	LOX	LH_2
Volume, ft^3	4620	11 650	357	902
Pressurization Band, psia	35-40	35-40	35-40	35-40
Vent Pressure Band, psia	41-46	41-46	41-46	41-46

PROPELLANT TANKAGE VEHICLE B — ORBITER

FIGURE A-29

		LOX	LH_2
Mass	lbs	2770	310
Liquid by Mass	—	21%	10%
Temperature of Vapor	°R	300	300
Total Pressure	psia	41	41
Heat Leak	$Btu/hr/ft^2$	7-36	4-TBD
Helium Pressurant used Main Tanks Only	lbs	2	23

MAIN TANK RESIDUAL PROPELLANTS UPSTREAM OF MAIN ENGINE VALVES AT ORBIT INSERTION VEHICLE B – ORBITER

ASCENT			
Station	Position	Q, $Btu/ft^2/sec$	TW °R
160	Base Region	6	2000

EXTERNAL SURFACE HEATING ENVIRONMENT
VEHICLE B – ORBITER

FIGURE A–30

ENTRY

Station*, ft	Position**, deg	Q, Btu/ft^2/sec	T_W, °R
36.4	115/245	1.3	1360
46.6	110/250	0.8	1200
	150/210	5.7	1960
	120/240	1.3	1360
160	Base Region	2.4	1570

* Station — Distance Along X-Axis, ft

** Position — Angular Position looking in the +X direction. Ref. zero degrees at top.

EXTERNAL SURFACE HEATING ENVIRONMENT
VEHICLE B — ORBITER

~ 500 °R Nominal After Long Soak in Orbit

~ 700 °R for 1 Hour During Reentry

INTERNAL HEATING ENVIRONMENT
VEHICLE B — ORBITER

FIGURE A-31

APPENDIX B

CONFIGURATION ANALYSES

B-1. CONFIGURATION ANALYSIS

Since low pressure APS engine weight and performance are sensitive to thrust level, chamber pressure, and expansion ratio, space shuttle APS configuration analyses were conducted to establish APS engine locations compatible with vehicle geometry constraints, and providing the best compromise between control moment arms (thrust level), and propellant feedline lengths (chamber pressure). These analyses enabled a realistic assessment of engine weight and performance based on available subsystem pressure budget and engine length, thus permitting valid subsystem weight definitions for comparison of candidate APS concepts. Presented in the following paragraphs are a definition of APS installations selected for the low pressure APS Subtask A studies, a brief description of engine operational logic, and results of engine/shuttle vehicle installation studies which established engine expansion ratio (length) constraints.

B-2. APS INSTALLATIONS

APS installations selected for subtask A study are illustrated in Figures B-1 and B-2 for orbiters A and B, and in Figures B-3 and B-4 for boosters A and B. As shown in Figure B-1, orbiter A APS engines were installed in four modules to take advantage of maximum available moment arms. Fore and aft modules provided both pitch and yaw attitude control, as well as Y and Z axis translation. Wing mounted modules provided both roll and X axis translation. In order to minimize propellant distribution network weight, engine feedlines were manifolded to existing main engine pressurization lines. In the orbiter B installation (Figure B-2), engines were grouped in fore and aft engine modules only. Wing-tip modules for this vehicle configuration were not attractive because of excessive oxygen feedline lengths. To conserve feedline pressure drop, APS engines for both boosters were grouped in fore-body modules as shown in Figures B-3 and B-4. As for orbiter A, engine feedlines for orbiter B and both boosters were manifolded to existing main engine pressurization lines to minimize feedline weight. Available control moment arms for these engine module locations are defined in Figures B-5 through B-8.

Applying available moment arms, total engine thrust levels were calculated consistent with translational and angular acceleration requirements tabulated in Figures B-9 and B-10 for the orbiters and boosters, respectively. Individual engine thrust levels were the same in all axes to minimize development requirements. In every case, nominal minimum acceleration levels were achieved with all engines firing and minimum acceleration levels were provided with two engines failed. This is illustrated by the engine operational logic presented in Figures B-11 through B-14.

Optimum APS engine expansion ratios were determined from subsystem design and weight sensitivity studies presented in Appendix F. Optimum engine expansion ratio for both boosters was 2.0. For the orbiter low velocity missions (+X axis maneuvers $\leq$ 10 ft/sec) optimum expansion ratio was 8.0; however, for intermediate and high velocity missions (+X axis maneuvers $\leq$ 50 ft/sec, and all +X axis maneuvers), an expansion ratio of 10.0 was found to be optimum. These values were compared against vehicle and tankage mold-lines to assess compatibility with available envelopes. Results are shown by the engine length/expansion ratio plot of Figure B-15. As shown, yaw axis control engines of orbiter B, and roll axis engines of orbiter A, were restricted to less than optimum expansion ratios.


APS FEED LINES
HELIUM PRESSURIZATION TANK
MAIN ENGINES LO2 FEED LINES
APS LH2 TANK
LO2 TANK PRESSURIZATION LINE
LO2 TANK
STA 60
STA 0.0
STA 374
STA 1736
MAIN ENGINES
LH2 TANK
MAIN ENGINE LH2 FEED LINES
APS FEED LINE
PAYLOAD
APS FEED LINES
ROTATED 90° CCW
ROTATED 90° CW
STA 60.0
MAIN ENGINE LO2 FEED LINE
APS LH2 TANK
(LO2 TANK)
C.G. @ ORBITAL INSERTION
MCDONNELL DOUGLAS CORPORATION
SKETCH - ORBITER "A" A.P.S. LINE INSTALLATION
LAO-001

FIGURE B-1

		MCDONNELL DOUGLAS ASTRONAUTICS COMPANY EASTERN DIVISION SAINT LOUIS, MISSOURI MCDONNELL DOUGLAS CORPORATION		
DR J. Gray	8-11-70	SKETCH - ORBITER "B" A.P.S. LINE INSTALLATION		
CHK				
STR				
GR				
		SIZE	CODE IDENT NO. 76301	LBO-001
		SCALE 1" = 10'	WT LB	SHEET

U

FIGURE B-2

FIGURE B-2

B-4


LO2 TANK
BOOST PROPULSION SYSTEM-LO2 TANK PRESSURIZATION LINE
LH2 TANK
THRUSTERS-AUXILIARY PROPULSION SYSTEM
AUXILIARY PROPULSION SYS.-H2 DISTRIBUTION LINE
BOOST PROPULSION SYS.-LH2 TANK PRESSURIZATION LINE
LO2 TANK PRESSURIZATION GAS - FROM MAIN ENGINES
ROTATED 90° CW
MCDONNELL DOUGLAS ASTRONAUTICS COMPANY
EASTERN DIVISION
SAINT LOUIS, MISSOURI
MCDONNELL DOUGLAS CORPORATION
DR J. Gray 6-19-70
CHK
STR
GR
BOOSTER-VEHICLE "A", INSTALLATION-AUXILIARY PROPULSION SYSTEM-LOW PRESSURE SYSTEM
SIZE
CODE IDENT NO. 76301
LAB-001
SCALE 1"=5'
WT
LB
SHEET

FIGURE B-3

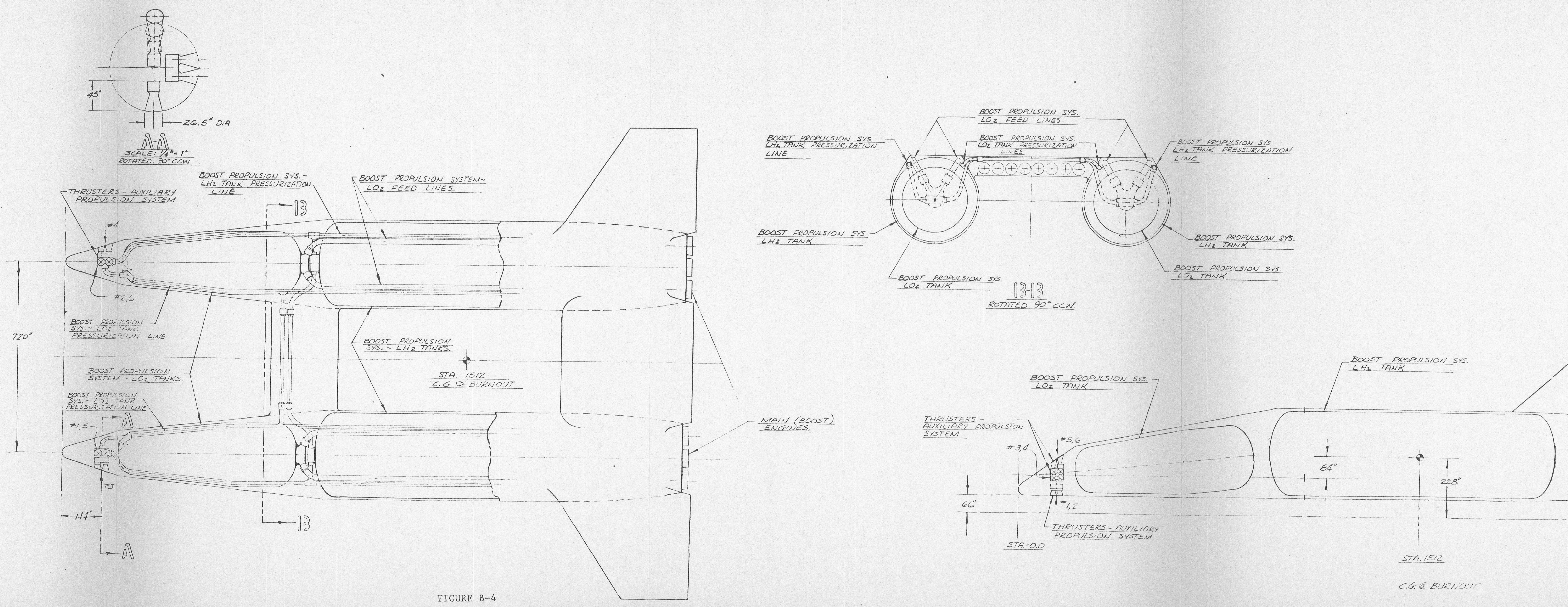
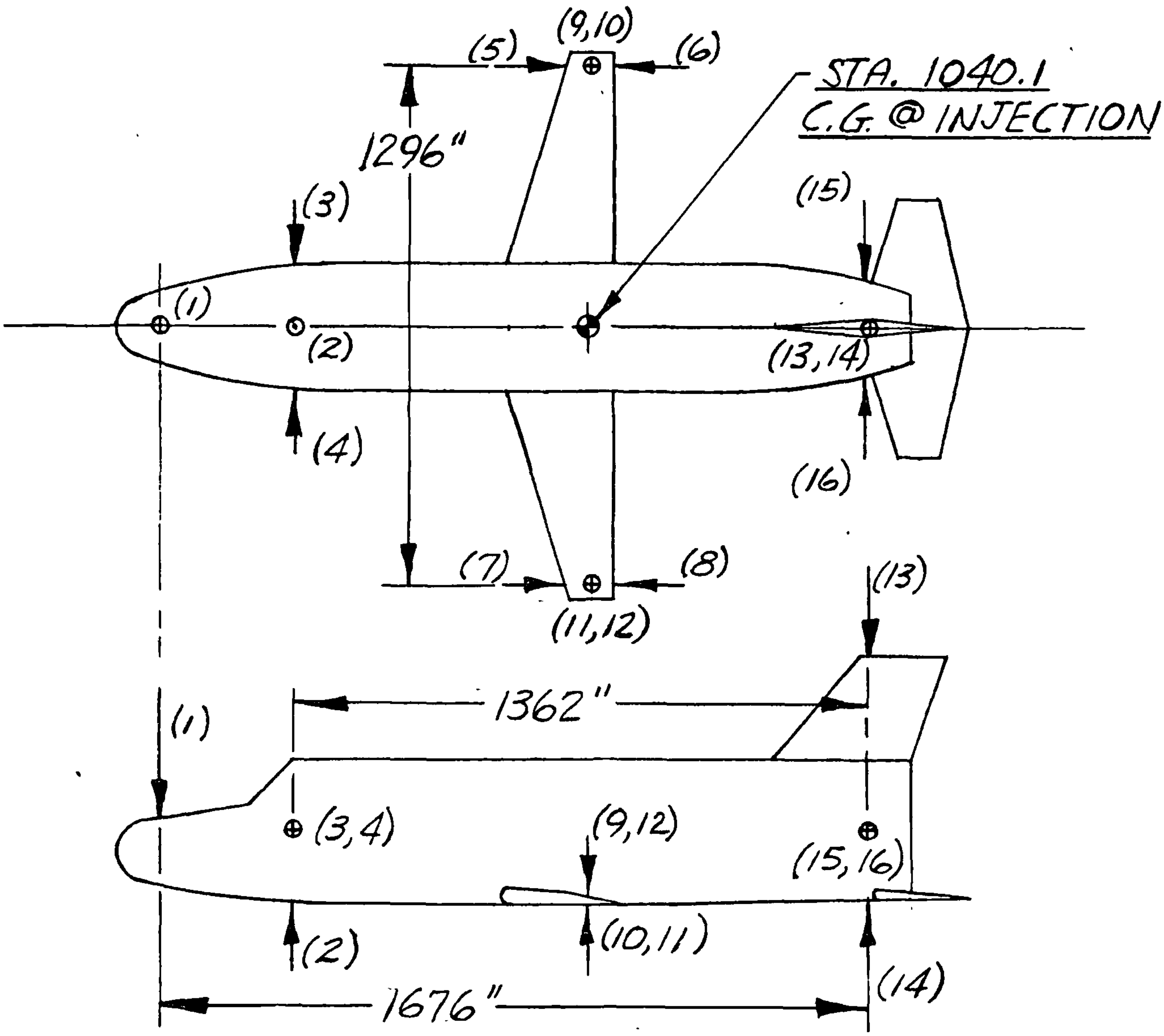


FIGURE B-4

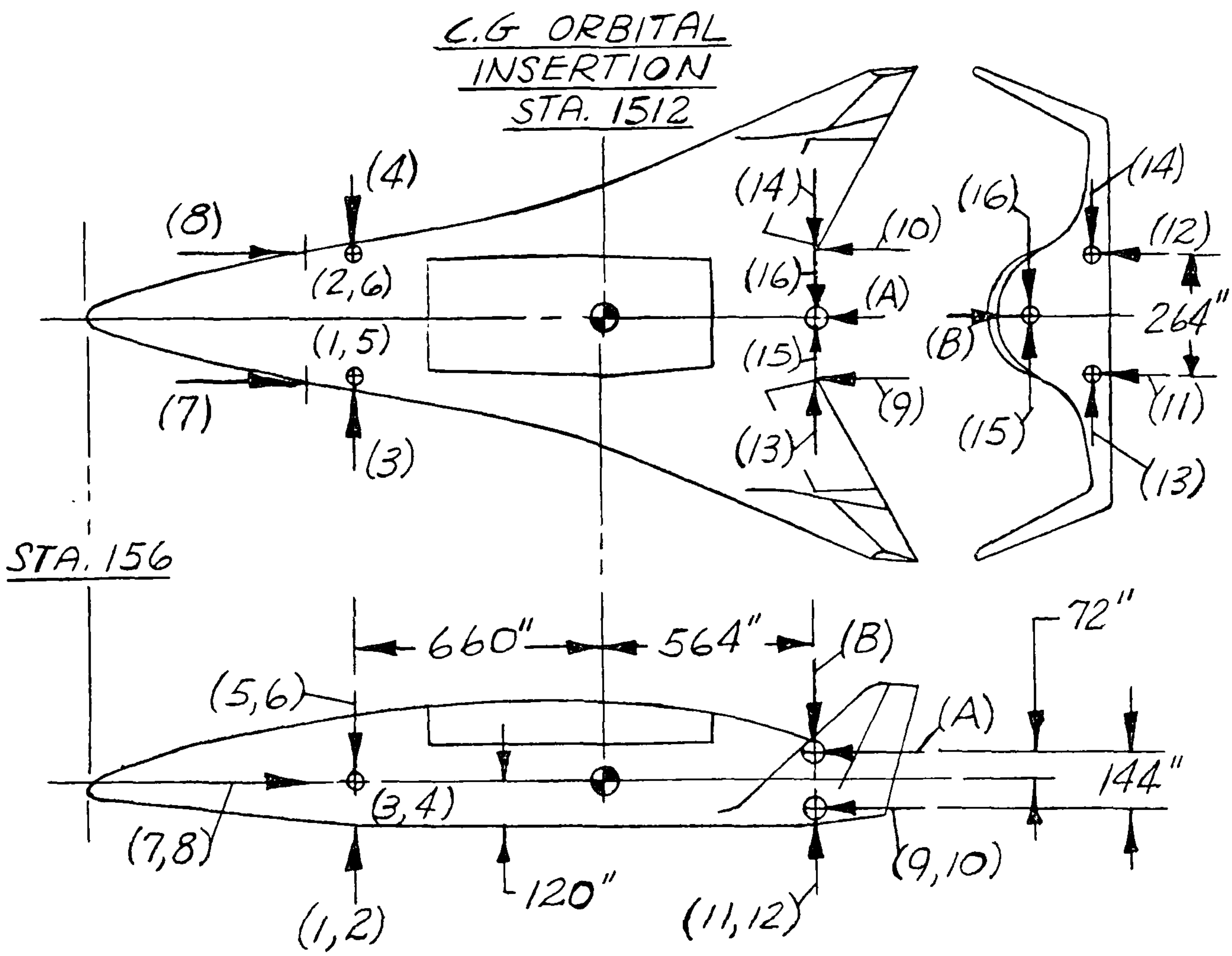
⚠ SCALE : 1/10 INCH = 1 FT. & NOTED

DR J. GRAY	MCDONNELL DOUGLAS ASTRONAUTICS COMPANY EASTERN DIVISION SAINT LOUIS, MISSOURI MCDONNELL DOUGLAS CORPORATION
CHK	INSTALLATION - LOW PRESSURE AUXILIARY PROPULSION SYSTEM - BOOSTER - VEHICLE "B"
STR	SIZE — CODE IDENT NO. 76301 — LB3-001
GR	SCALE ⚠ — WT — LB — SHEET 1 OF 1



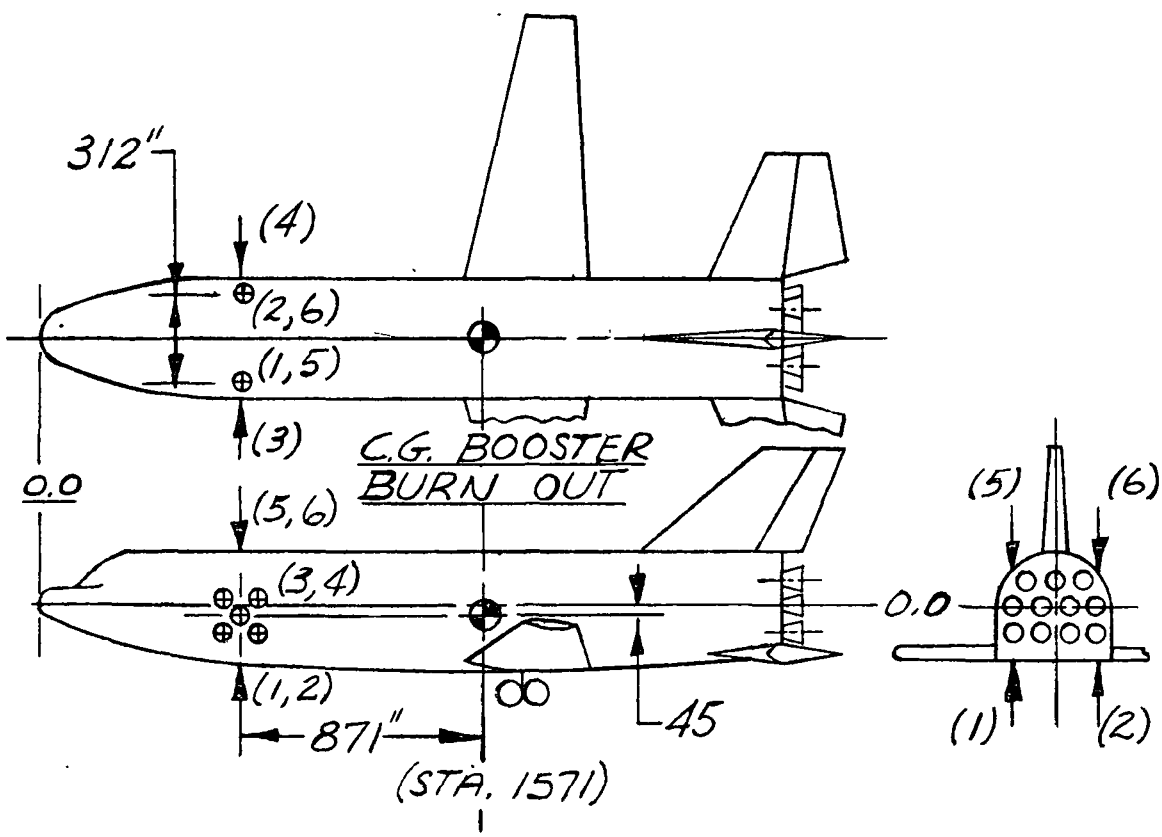
THRUSTER LOCATIONS - VEHICLE A - ORBITER

FIGURE B-5



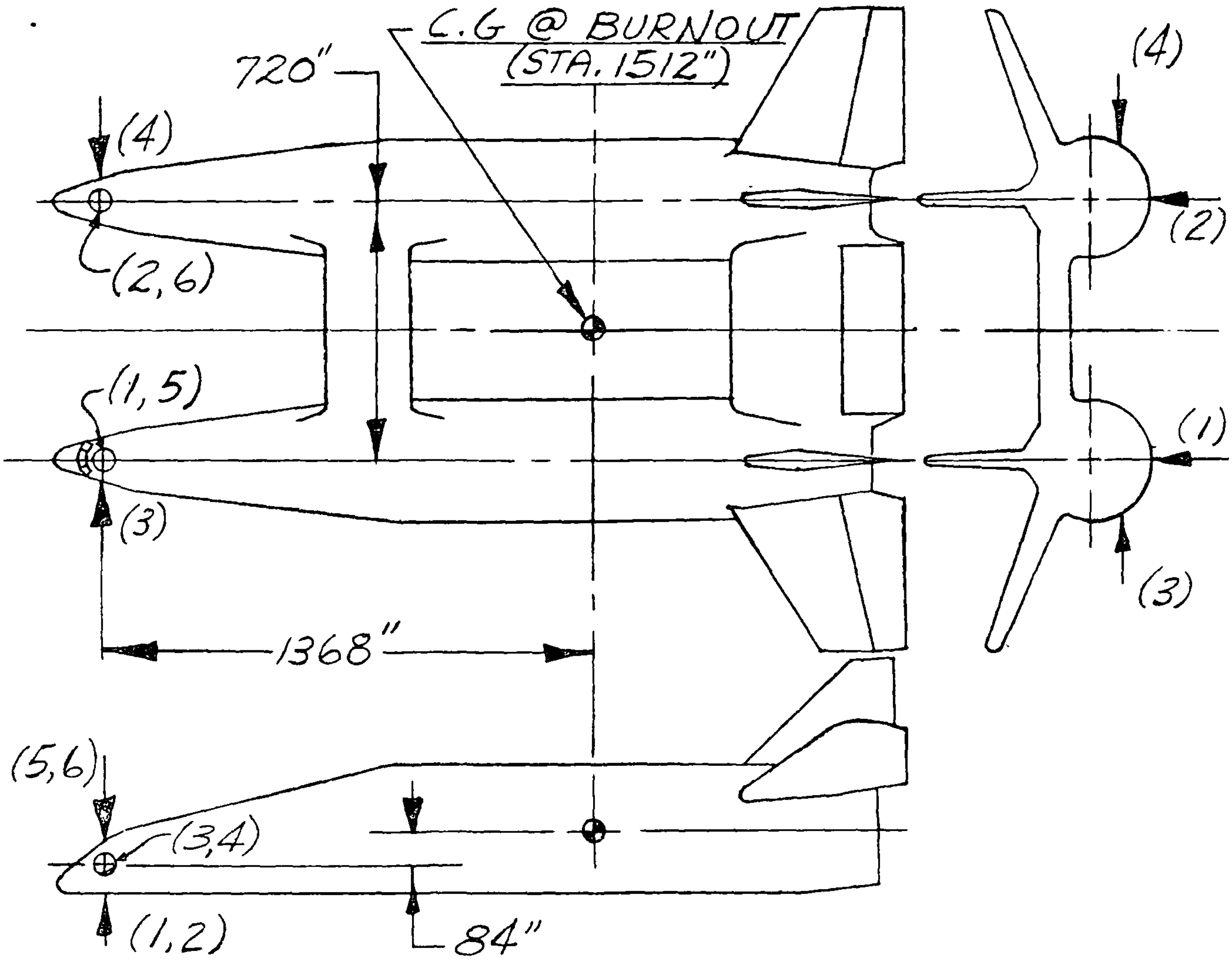
THRUSTER LOCATIONS - VEHICLE B - ORBITER

FIGURE B-6



THRUSTER LOCATIONS- VEHICLE A- BOOSTER

FIGURE B-7



THRUSTER LOCATIONS - VEHICLE B - BOOSTER

FIGURE B-8

	ACCELERATION					
	(FT/SEC^2)			(DEG/SEC^2)		
	X	Y	Z	PITCH	YAW	ROLL
MINIMUM	.07	.07	.07	.3	.3	.3
NOMINAL MINIMUM	.1	.1	.1	.5	.5	.5

- TOTAL THRUST LEVEL PER AXIS CALCULATED TO PROVIDE ON-ORBIT REQUIREMENTS
- NUMBER OF ENGINES AND THRUST LEVEL CALCULATED TO PROVIDE:
  - NOMINAL MINIMUM ACCELERATION WITH ALL ENGINES FIRING
  - MINIMUM ACCELERATION WITH TWO ENGINES OUT
- MINIMUM NUMBER OF ENGINES SELECTED (MINIMUM WEIGHT).

GROUND RULES FOR ORBITER ENGINE SIZING

FIGURE B-9

	ACCELERATION (DEG/SEC^2)		
	PITCH	YAW	ROLL
MINIMUM NOMINAL MINIMUM	.3 .5	.3 1.0	.3 1.0

- TOTAL THRUST LEVEL PER AXIS CALCULATED TO MEET REENTRY REQUIREMENTS
- NUMBER OF ENGINES AND THRUST LEVEL CALCULATED TO PROVIDE:
  - NOMINAL MINIMUM ACCELERATION WITH ALL ENGINES FIRING
  - MINIMUM ACCELERATION WITH TWO ENGINES OUT
- BEST COMPROMISE OF NUMBER OF ENGINES AND THRUST LEVEL SELECTED (MINIMUM WEIGHT).

GROUND RULES FOR BOOSTER ENGINE SIZING

FIGURE B-10

MANEUVER	ENGINE ASSEMBLIES	OPERATIONAL MODE	ENGINES USED	ACCELERATION LEVEL (FT/SEC^2 OR DEG/SEC^2)	COUPLING EFFECTS
+X	6 & 8	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	6a,8a 6b,8b 6b	.143 .143 .0715	PITCH PITCH PITCH,YAW
-X	5 & 7	SAME AS +X			
+Y	4 & 16	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	4a,16a 4b,16b 4b	.143 .143 .0715	NONE NONE YAW
-Y	3 & 15	SAME AS +Y			
+Z	2 & 14	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	2a,14a 2b,14b 2b	.143 .143 .0715	NONE NONE PITCH
-Z	1 & 13	SAME AS +Z			(YAW)
+PITCH	2 & 13	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	2a,2b,13a,13b 2b,13b 2a,2b	.621 .310 .310	NONE NONE Z
-PITCH	1 & 14	SAME AS + PITCH			(YAW)
+YAW	4 & 15	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	4a,4b,15a,15b 4b,15b 4a,4b	.620 .310 .310	NONE NONE Y
-YAW	3 & 16	SAME AS +YAW			
+ROLL	9 & 11	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	9a,11a 9b,11b 9b	2.98 2.98 1.49	NONE NONE Z
-ROLL	10 & 12	SAME AS + ROLL			

ENGINE OPERATIONAL PROCEDURE

ORBITER A

FIGURE B-11

MANEUVER	ENGINE ASSEMBLIES	OPERATIONAL MODE	ENGINES USED	ACCELERATION LEVEL (FT/SEC^2 OR DEG/SEC^2)	COUPLING EFFECTS
+X	9,10, & A	NOMINAL	9,10,Aa,Ab	.500	NONE
		ONE ENGINE OUT	10,Aa	.250	YAW
		TWO ENGINES OUT	10,Aa	.250	YAW
-X	7 & 8	NOMINAL	7a,8a	.250	NONE
		ONE ENGINE OUT	7b,8b	.250	NONE
		TWO ENGINES OUT	7b	.125	YAW
+Y	3,13 & 15	NOMINAL	3a,13a,15a	.375	NONE
		ONE ENGINE OUT	3b,13b,15b	.375	NONE
		TWO ENGINES OUT	3b,13b	.250	ROLL
-Y	4,14 & 16	SAME AS +Y			
+Z	1,2,11 & 12	NOMINAL	1,2,11,12	.500	NONE
		ONE ENGINE OUT	1,12	.250	NONE
		TWO ENGINES OUT	1,(OR 1,2)	.125	PITCH,ROLL
-Z	5,6 & B	NOMINAL	5,6,Ba,Bb	.500	NONE
		ONE ENGINE OUT	6,Ba,Bb	.375	ROLL
		TWO ENGINES OUT	6,Ba	.250	ROLL
+PITCH	1,2 & B	NOMINAL	1,2,Ba,Bb	.970	NONE
		ONE ENGINE OUT	2,Ba	.485	ROLL
		TWO ENGINES OUT	Ba,Bb	.485	Z
-PITCH	5,6,11 & 12	NOMINAL	5,6,11,12	.970	NONE
		ONE ENGINE OUT	5,12	.485	NONE
		TWO ENGINES OUT	5,11	.485	ROLL
+YAW	3,14 & 16	NOMINAL	3a,3b,14a,16a	.875	NONE
		ONE ENGINE OUT	3b,14a,16a	.656	NONE
		TWO ENGINES OUT	14a,16a	.438	Y
-YAW	4,13 & 15	SAME AS +YAW			
+ROLL	14 & 15 (1 & 6)	NOMINAL	14a,14b,15a,15b	.603	NONE
		ONE ENGINE OUT	14b,15b	.3015	NONE
		TWO ENGINES OUT	1,6	.552	NONE
-ROLL	13 & 16 (2 & 5)	SAME AS +ROLL			

ORBITER B ENGINE OPERATIONAL PROCEDURE

FIGURE B-12

MANEUVER	ENGINE ASSEMBLIES	OPERATIONAL MODE	ENGINES USED	ACCELERATION LEVEL (DEG/SEC^2)	COUPLING* EFFECTS
+ PITCH	1 & 2	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	1a,1b,2a,2b 1b,2b 1a,1b	.814 .412 .412	NONE NONE ROLL
- PITCH	5 & 6	SAME AS + PITCH			
+ YAW	3	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	3a,3b,3c,3d,3e 3b,3c,3e 3c,3d,3e	1.02 .612 .612	NONE NONE ROLL
- YAW	4	SAME AS + YAW			
+ ROLL	1 & 6 (3 & 4)	NOMINAL ONE ENGINE OUT TWO ENGINES OUT	1a,1b,6a,6b 1b,6b 3a,4c	1.900 .950 .950	NONE NONE NONE
- ROLL	2 & 5 (3 & 4)	SAME AS + ROLL			

* TRANSLATION EFFECTS NOT CONSIDERED

ENGINE OPERATIONAL PROCEDURE

BOOSTER A

FIGURE B-13

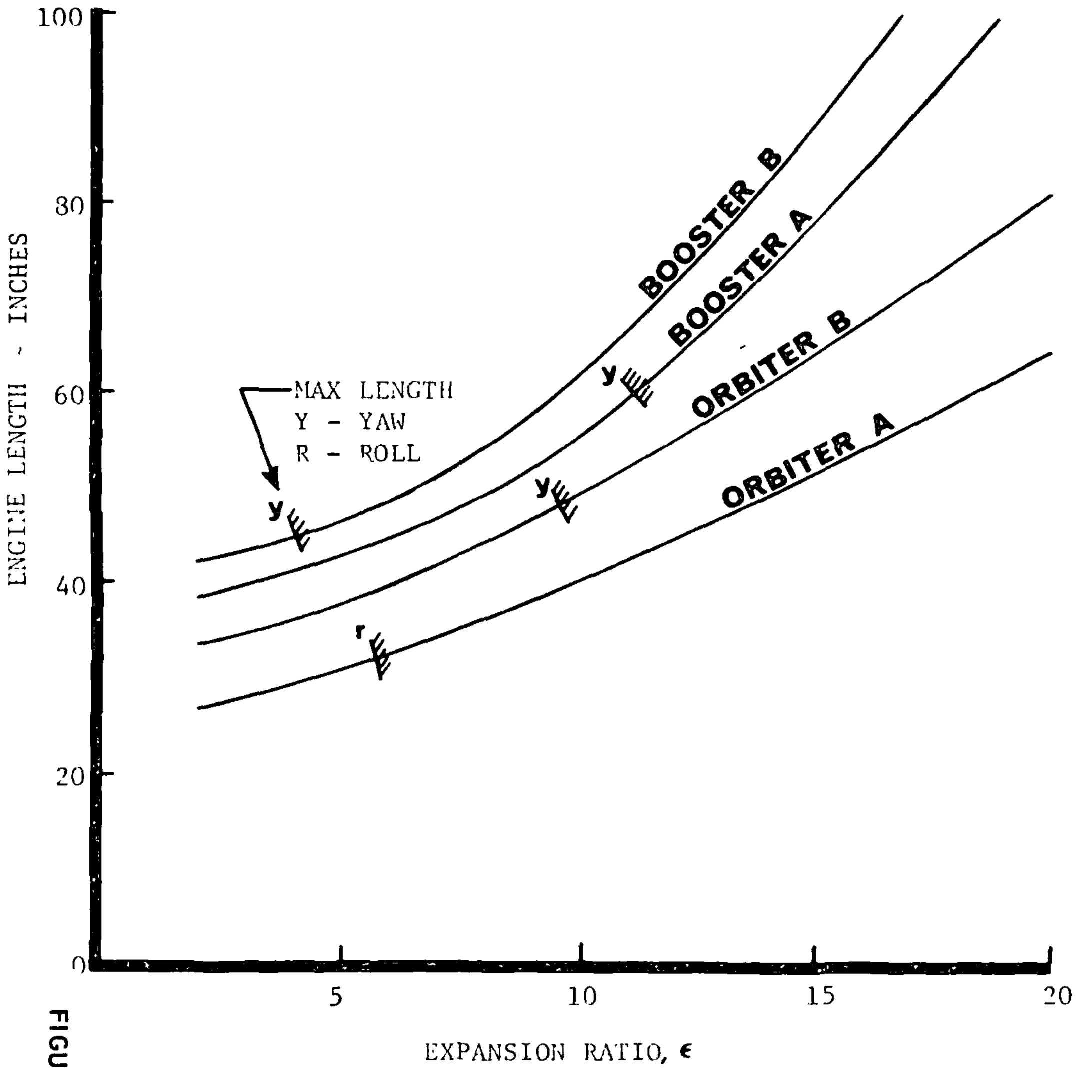
MANEUVER	ENGINE ASSEMBLIES	OPERATIONAL MODE	ENGINE USED	ACCELERATION LEVEL (DEG/SEC^2)	COUPLING * EFFECTS
+ PITCH	1 & 2	NOMINAL	1a,2a	.756	NONE
		ONE ENGINE OUT	1b,2b	.756	NONE
		TWO ENGINES OUT	2b	.378	ROLL
- PITCH	5 & 6	SAME AS + PITCH			
+ YAW	3	NOMINAL	3a,3b,3c,3d	1.155	ROLL
		ONE ENGINE OUT	3b,3c,3d	.865	ROLL
		TWO ENGINES OUT	3c,3d	.578	ROLL
- YAW	4	SAME AS + YAW			
+ ROLL	1 & 6	NOMINAL	1a,1b,6a,6b	1.03	NONE
		ONE ENGINE OUT	1b,6b	.515	NONE
		TWO ENGINES OUT	6a,6b	.515	PITCH
- ROLL	2 & 5	SAME AS + ROLL			

* TRANSLATION EFFECTS NOT CONSIDERED

ENGINE OPERATIONAL PROCEDURE

BOOSTER B

FIGURE B-14



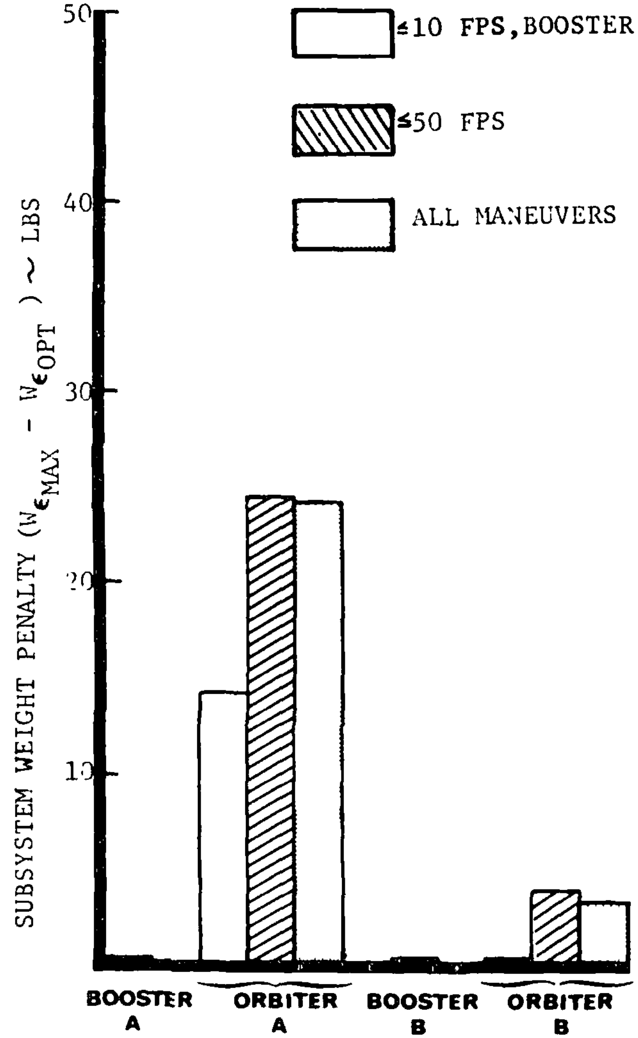


FIGURE B-15

ENGINE ENVELOPE CONSTRAINTS

Subsystem weight penalties associated with these expansion ratio restrictions were minor as shown by the bar graph of Figure B-15. For the remaining orbiter control axes and both boosters, optimum engine expansion ratios were compatible with vehicle and tankage envelopes.

Using results of the analyses presented above, realistic engine weights and performance levels were determined for each of the space shuttle boosters and orbiters, applying the data of Appendix D. These engine weights, and impulsive propellant requirements calculated using the delivered performance levels, were employed for quantitative comparisons of candidate low pressure APS concepts.

APPENDIX C

APS CONCEPT SCREENING STUDIES

C-1. INTRODUCTION

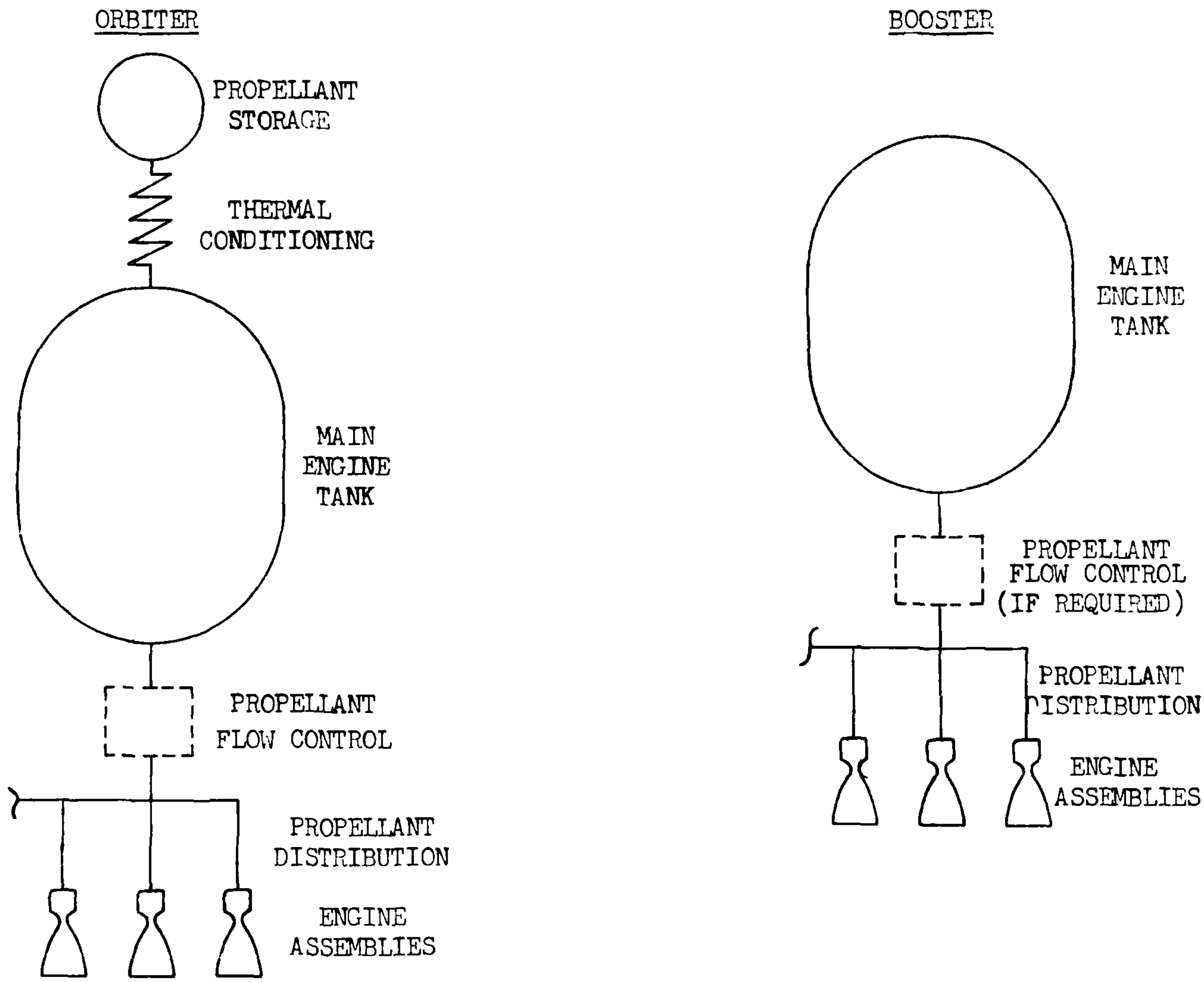
Preliminary mission simulations for space shuttle orbiter and booster APS yielded three fundamental results:

(1) To satisfy orbiter APS total impulse requirements, supplementary propellant, in addition to residual propellant left in main engine tanks following boost, would be required.

(2) To maintain acceptable propellant feed pressures and temperatures in the orbiter APS, thermal conditioning of resupply propellant injected into the main engine tanks would be required except for APS with very low total impulse requirements.

(3) To satisfy booster APS mission total impulse required use of boost residual propellant vapor, only, and supplementary APS propellant would not be required.

These results enabled definition of basic assemblies required for a low pressure APS as illustrated in Figure C-1 for the orbiters and boosters. At the start of the Subtask A study, various design alternatives were considered for each of the required subsystem assemblies (propellant storage, thermal conditioning, and propellant flow control). Alternate assembly concepts were screened to identify those high-value assembly concepts deserving further analysis and comparison. Presented in this appendix are a description of the various assembly concepts considered, and the screening studies performed to reduce the number of candidate assembly concepts.



FUNDAMENTAL LOW PRESSURE APS CONCEPTS

FIGURE C-1

C-2. ASSEMBLY DESIGN ALTERNATIVES

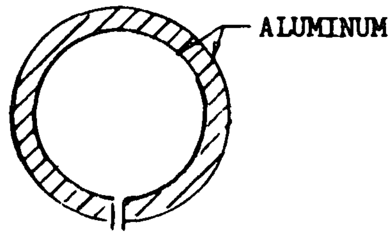
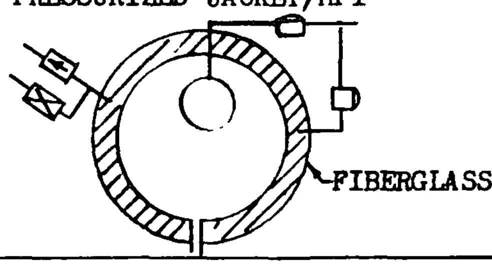
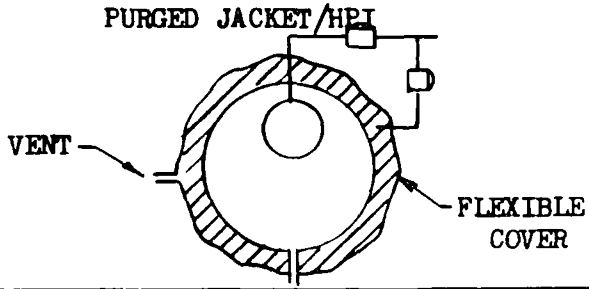
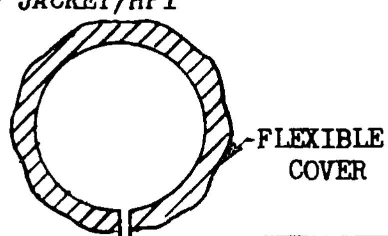
Various design alternatives were considered for orbiter APS propellant storage, thermal conditioning, and propellant flow control functions. The booster APS, on the other hand, did not require main engine tank propellant resupply so only flow control design alternatives had to be considered. Analyses performed to compare various assembly design alternatives, and selections for further study, are described below.

C-2.1 Orbiter Propellant Storage - Design alternatives for propellant storage state, propellant pressurization, and APS/OMS propellant integration were investigated and compared.

Design Alternatives - Both liquid and supercritical propellant storage assemblies were considered. Liquid storage was attractive because it offered light weight tankage, but had the disadvantage of a required propellant positioning assembly. Propellant storage in the supercritical state eliminated the need for positioning, but because of higher operating pressures, required significantly heavier tanks. To compare the two storage approaches it was necessary to establish weight and design models that would allow a realistic assessment of their merit. The design alternatives together with the rationale leading to selection of the tank models are provided in the following paragraphs.

Various liquid tankage concepts are illustrated in Figure C-2. Each concept employs an aluminum storage vessel protected by high performance insulation (HPI) to minimize liquid vaporization and associated propellant vent losses. A screen surface tension device is employed for propellant acquisition. The HPI is sealed using either a flexible or hard outer cover to prevent moisture condensation and/or cryopumping of HPI during ground operations. Cooling tubes are contained between the cover and tank walls to absorb incoming heat. In operation, a small amount of propellant is bled from the storage tank, throttled to saturation condition and passed through the cooling tubes where incoming heat is absorbed through vaporization.

Concepts employing flexible covers provided minimum weight but greatest susceptibility to insulation crushing from multiple pressure cycling or handling loads. Concepts employing hard outer covers minimized risk of insulation damage, but were heavier, especially for an aluminum outer shell. The liquid tankage concept selected for later tankage screening and integration studies employed a fiberglass outer shell, which was pressurized during ascent to orbit, evacuated

TANKAGE/HPI CONCEPTS	RELATIVE WEIGHT RATIO	FEATURES
VACUUM JACKET/HPI 	1.74 - 1.85	° REUSABLE HPI WITH ALUMINUM COVER ° VACUUM JACKETED PRESSURE VESSEL ° ALUMINUM INNER AND OUTER SHELL ° COLLAPSE PRESSURES RESULT IN HEAVY OUTER SHELL ° NO PURGE OF HPI REQUIRED
✓ PRESSURIZED JACKET/HPI 	1.38 - 1.46	° REUSABLE HPI WITH FIBERGLASS COVER ° HPI PRESSURIZED DURING DESCENT AND ASCENT ° PRESSURIZED FIBERGLASS OUTER SHELL RESULTS IN LOWER WEIGHT ° INCREASED COMPLEXITY DUE TO VENT/PRESSURIZATION SYSTEM
PURGED JACKET/HPI 	≈1.0	° REUSABLE HPI WITH FLEXIBLE COVER ° HPI VENTED TO ATMOSPHERE AND PURGED DURING ASCENT AND DESCENT ° LOW STRUCTURAL WEIGHT
SEALED JACKET/HPI 	1.0 *	° REUSABLE HPI WITH FLEXIBLE COVER ° HPI EVACUATED AND SEALED ° LOWEST WEIGHT CONCEPT ° HIGHEST TECHNOLOGY RISK DUE TO POSSIBLE HPI CRUSHING AND DEVELOPMENT OF SEALING METHOD

✓ SELECTED * H_2 AND O_2 DRY TANKAGE WEIGHT = 389 LB

FIGURE C-2

TANKAGE/HPI CONCEPTS

on orbit, and repressurized during reentry to minimize shell weight. This concept provided the best compromise between weight and development risk. (Corresponding oxygen and hydrogen tank weights for this concept are given in Figure C-3 as a function of both propellant weight and tank pressure.) Two liquid tank pressurization concepts (illustrated in Figure C-4) were considered in the later screening and integration studies: conventional cold gas helium pressurization, with helium bottles installed in the propellant tanks to reduce bottle size; and autogeneous pressurization, wherein warm propellant gas is tapped from main engine tanks and compressed to required ullage pressure.

Supercritical propellant storage was also attractive because propellant is stored in a single phase eliminating need for a propellant acquisition device. A typical storage concept is analyzed in Figure C-5. (The HPI thermal protection concept is the same as that selected for liquid storage.) Propellant is stored initially at pressures above the critical point (188 lb $/in^2$a for hydrogen and 730 lb $/in^2$a for oxygen) at low temperature (high density) to conserve tank volume. As propellant is withdrawn, energy is added to remaining propellant to sustain tank pressure. This, in turn, increases the temperature (reduces density) of the remaining fluid throughout the expulsion process. A typical expulsion process is illustrated on the oxygen pressure-enthalpy diagram of Figure C-5. The low final propellant density attained at end of expulsion minimizes the amount of residual propellant, promoting good propellant expulsion efficiency. In the example of Figure C-5, energy for heating remaining propellant is derived from hot gas generator exhaust gases. Since gas generator propellant demands increase with rising final propellant tank temperatures, a trade-off between gas generator propellant requirements and tank residual propellant was conducted to determine optimum final tank temperatures. Results, shown in Figure C-5, provided minimum propellant losses at final tank temperatures of 100 R and 500 R for hydrogen and oxygen, respectively.

Various alternatives for adding energy to maintain supercritical tank pressures are illustrated and compared in Figure C-6. Each approach uses an active heat exchanger/gas generator assembly to provide required thermal energy. Concepts using an in-tank heat exchanger were particularly heavy. This result was caused by low convective heat transfer coefficients between the heat exchanger and supercritical fluid, which necessitated large heat exchanger surface areas. Better thermal efficiency and lower assembly weights were achieved with external heat exchangers, using pumps to provide high fluid velocities and correspondingly high

- INCLUDES MOUNTS
- FIBERGLASS OUTER SHELL

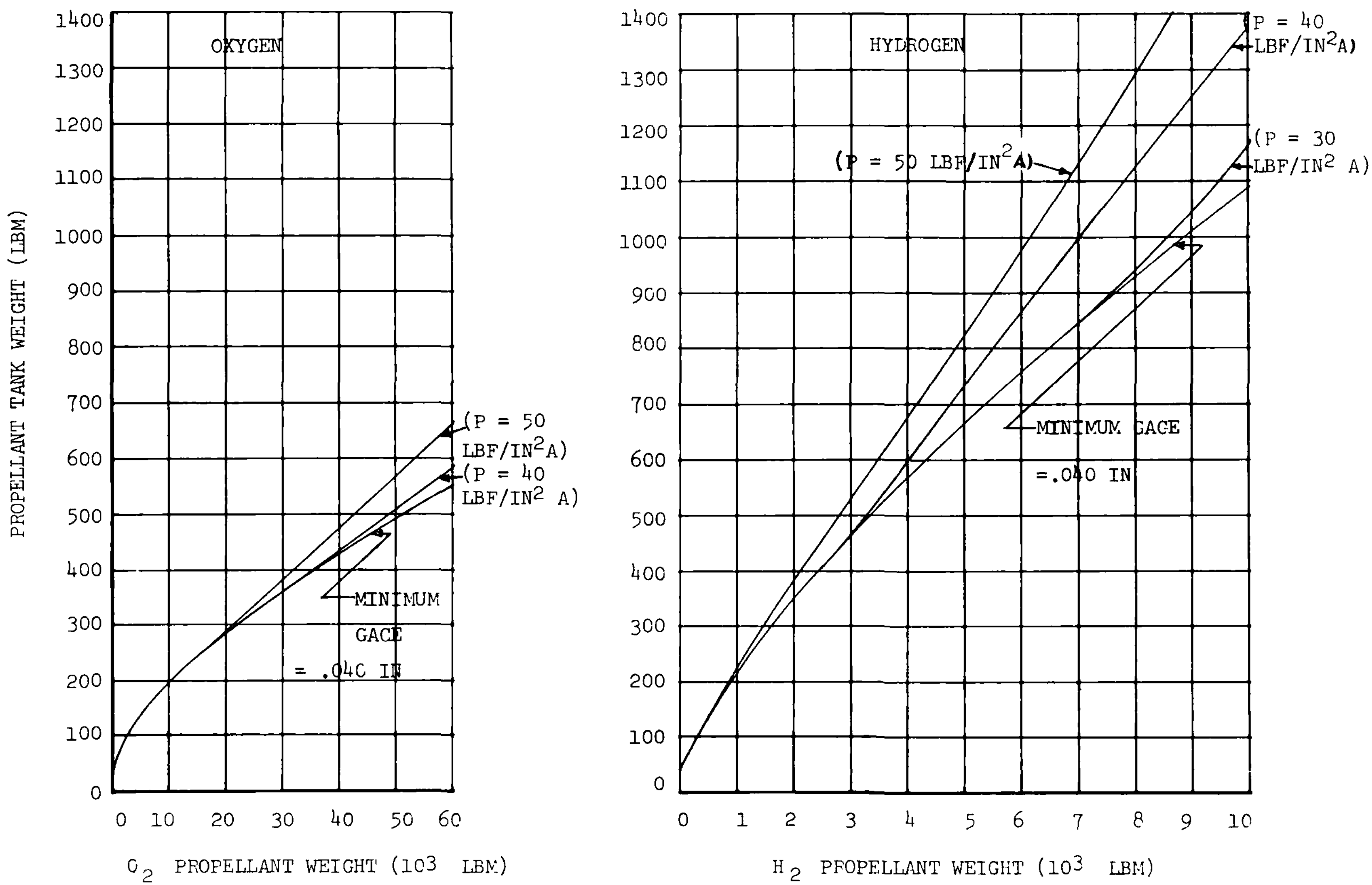


FIGURE C-3

PROPELLANT TANK WEIGHTS

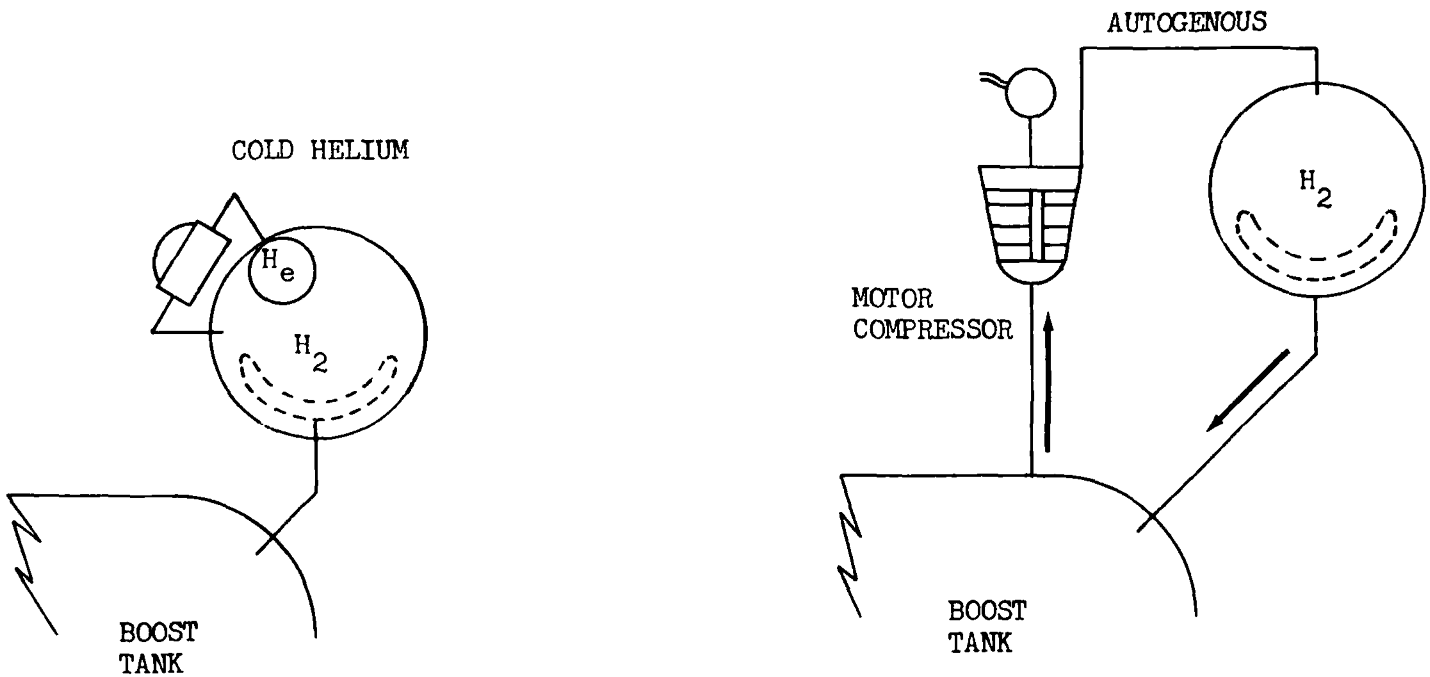


FIGURE C-4

LIQUID PROPELLANT PRESSURIZATION CONCEPTS

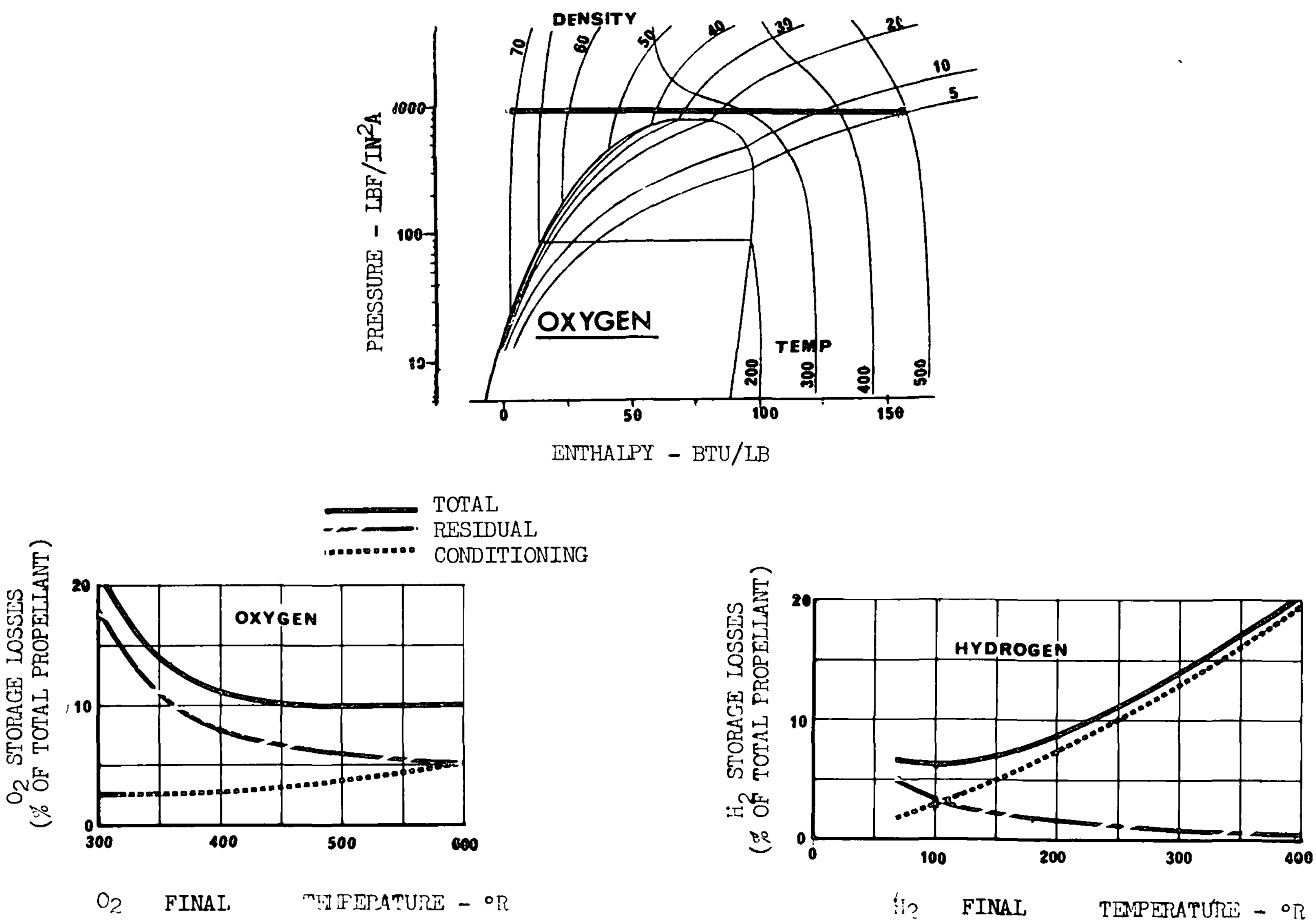
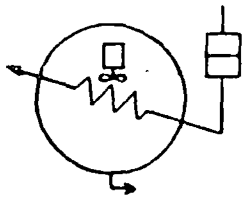
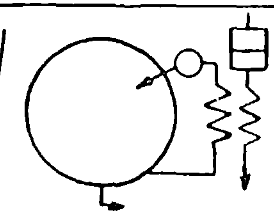
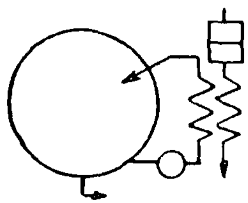
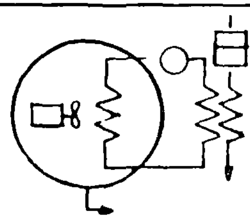
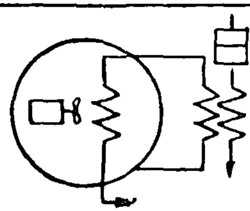


FIGURE C-5

SUPERCRITICAL PROPELLANT STORAGE CONCEPT

CONCEPT	DESIGN FEATURES	RELATIVE WEIGHT RATIO	COMMENTS
	IN TANK HEAT EXCHANGER - CIRCULATION DEVICE PROVIDES FORCED CONVECTION HEAT TRANSFER ON COLD SIDE	4.1	EXPLOSION HAZARD IN O_2 TANK REDUNDANCY DIFFICULT TO IMPLEMENT HOT GAS TUBING PENETRATION THROUGH TANK
√ 	EXTERNAL HEAT EXCHANGER - PUMP CIRCULATES FLUID THROUGH SHELL	1.8*	ISOLATION PROVIDES EXPLOSION PROTECTION REDUNDANCY PROVISIONS EASIER TO IMPLEMENT PUMP OPERATION WITH CONSTANT DENSITY FLUID
	EXTERNAL HEAT EXCHANGER - PUMP CIRCULATES FLUID THROUGH SHELL	1.0	ISOLATION PROVIDES EXPLOSION PROTECTION REDUNDANCY PROVISIONS EASIER TO IMPLEMENT PUMP OPERATION WITH VARIABLE DENSITY FLUID
	INTERMEDIATE HELIUM LOOP - IN TANK HEAT EXCHANGER WITH CIRCULATION DEVICE - EXTERNAL HEAT EXCHANGER WITH PUMP	28.5	NO EXPLOSION PROTECTION REQUIRED REDUNDANCY DIFFICULT TO IMPLEMENT
	OPEN LOOP SYSTEM - IN TANK HEAT EXCHANGER WITH CIRCULATION DEVICE - EXTERNAL HEAT EXCHANGER NO PUMP	47.5	ISOLATION PROVIDES EXPLOSION PROTECTION REDUNDANCY PROVISIONS EASIER TO IMPLEMENT

√ SELECTED * H_2 PUMP AND HEAT EXCHANGER WEIGHT = 19 LB

FIGURE C-6

SUPERCRITICAL STORAGE TANK HEAT EXCHANGER COMPARISON heat transfer coefficients. The concept selected for later screening and integration studies employs an external heat exchanger/gas generator circuit, in which auxiliary propellant is tapped from the supercritical storage tank, heated to prescribed temperature, pumped to a higher pressure, and cycled back into the storage vessel. This approach provided reasonable weight, eliminated any explosion hazard from gas generator exhaust products and allowed the pump to operate at constant fluid inlet density.

Various APS/OMS propellant integration options considered for both liquid and supercritical storage are shown in Figure C-7. Concepts considered for liquid storage included:

(1) fully integrated tankage with common APS/OMS propellant acquisition
(2) completely separate, nonrefillable APS tanks
(3) combined, nonrefillable tankage, with APS tanks mounted inside OMS tanks
(4) separate, refillable APS tanks to minimize tank size, and
(5) combined refillable APS tankage

Since the baseline OMS employed liquid propellant storage, only two integration concepts were considered for supercritical storage - separate nonrefillable tankage and separate refillable tankage in which liquid supplied from the OMS tank is pumped to supercritical storage pressures for APS refill.

<u>Concept Screening</u> - Design alternatives (presented above) were compared to define the most attractive APS/OMS integration options for both liquid and supercritical storage, and to identify the best liquid propellant pressurization scheme. Comparisons were made using shuttle A orbiter requirements tabulated in Figure C-8 for three distinct mission velocity levels:

(1) all translation maneuvers
(2) all translation maneuvers except those in the +X direction greater than 50 ft/sec
(3) all translation maneuvers except those in the +X direction greater than 10 ft/sec

As shown in Figure C-8, APS mission total impulse ranged from approximately 1M to 14M lb-sec, depending upon maneuver velocity allocation between APS and OMS. APS propellant tank sizes for refillable tankage concepts were based on a reentry total impulse requirement of 420,690 lb-sec, for the low velocity mission (+X axis maneuvers $\leq$10 ft/sec); and the maximum impulse between major burns (1,720,000 lb-sec) for the intermediate velocity mission (+X axis maneuvers $\leq$50 ft/sec). The minimum number of APS tank refills was determined by dividing total APS mission

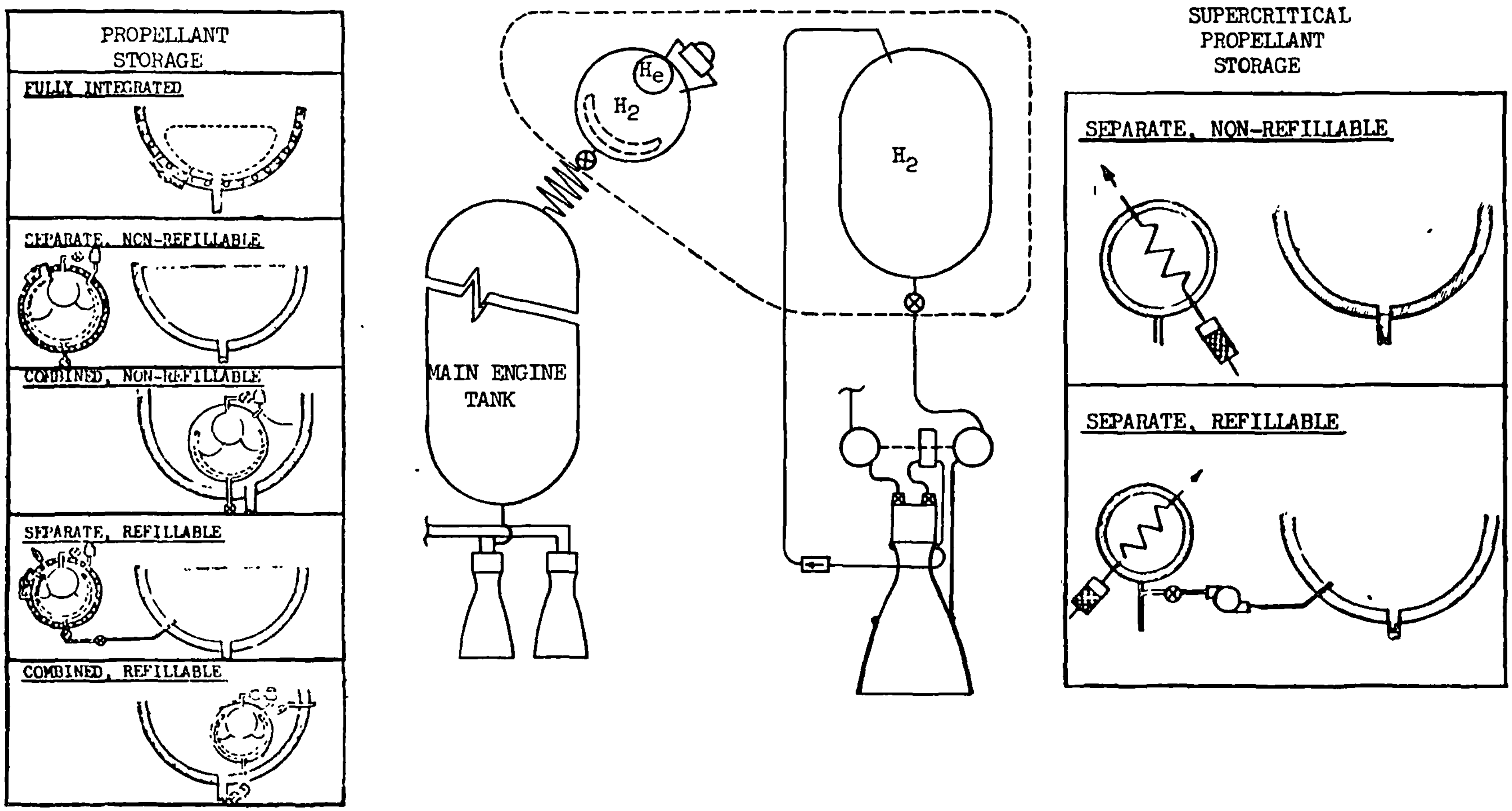


FIGURE C-7

APS/OMS PROPELLANT INTEGRATION CONCEPTS

VEHICLE A

VARIABLE \ MISSION	≤ 10 FPS		≤ 50 FPS		ALL MANEUVERS
	APS	OMS	APS	OMS	APS
IMPULSIVE TOTAL IMPULSE * (10^6 LB-SEC)	.960	12.978	2.775	11.163	13.938
NUMBER OMS STARTS	-	18	-	7	-
REFILLABLE APS - NO REFILLS	4	-	2	-	-
REFILLABLE APS - APS PROP.	1090	-	4453	-	-
PROPELLANT					
- IMPULSE	2,490	29,200	7,180	25,100	36,100
- START LOSSES	-	2,590	--	1,010	-
- OMS SETTLING	1,400	-	544	-	-
(TOTAL/SYS)	(3,890)	(31,790)	(7,724)	(26,110)	(36,100)
TOTAL USABLE	35,680		33,834		36,100

* TOTAL LESS SETTLINGS AND START LOSSES

FIGURE C-8

APS/OMS TANKAGE REQUIREMENTS

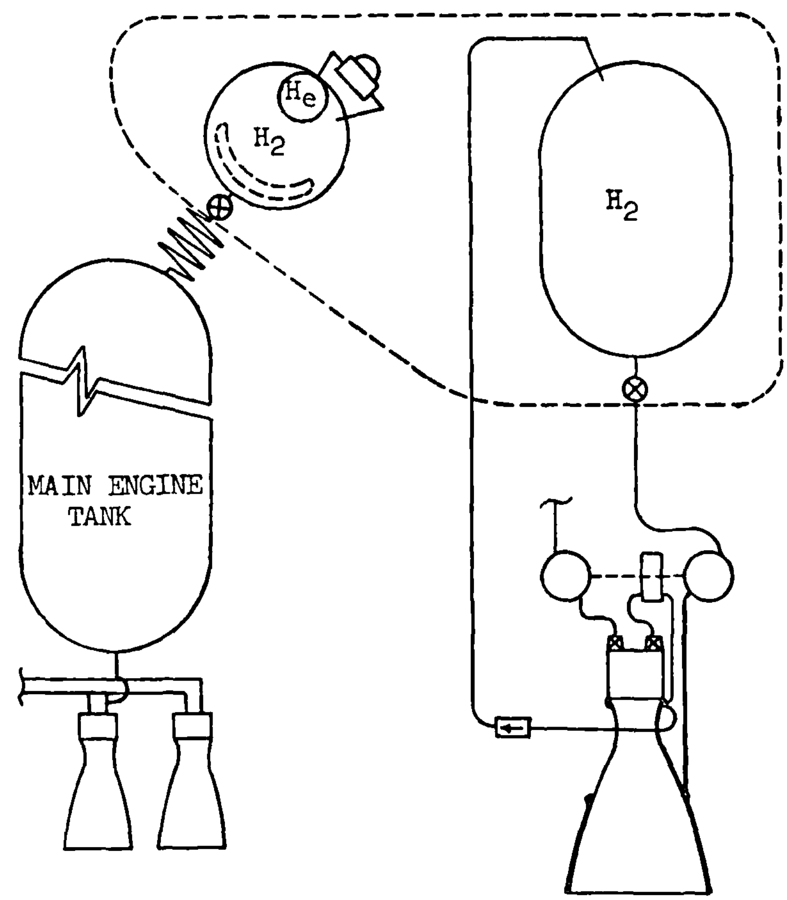
propellant weight by propellant weight required for these impulse levels. For separate and combined tankage configurations, APS total impulse included an allocation of 30,000 lb-sec per maneuver for settling liquid propellant in the OMS. OMS propellant requirements included 144 lb of propellant for each OMS engine start to fill engine supply lines. Assumed engine performance and nominal propellant tank operating pressures are provided in Figure C-9. OMS engine performance values were bases on the Pratt and Whitney RL10A3-3 engine.

To ensure fair and valid comparisons, storage concepts were assessed on the basis of weight and volume (25 points), mission flexibility (25 points), technology requirements (20 points), design simplicity (15 points), and development requirements (15 points). These criteria and weighting were further subdivided into the specific areas summarized in Figure C-10.

Total APS/OMS weights for liquid APS storage are compared in Figures C-11 and C-12 for oxygen and hydrogen, respectively. These weights include the effect of APS/OMS mixture ratio differences, which provided overall mission mixture ratios of 4.7 for low maneuver level mission (+X axis maneuvers $\leq$10 ft/sec), 4.4 for intermediate maneuver level mission (+X axis maneuvers $\leq$50 ft/sec), and 3.0 for high maneuver level mission. As shown in Figure C-11, weight variation among liquid oxygen storage concepts was very slight. However, from Figure C-12 it is seen that autogenous pressurization of liquid hydrogen tanks provided significant weight advantage.

Detailed ratings of alternate APS/OMS integration and pressurization options for liquid storage are shown in Figures C-13 through C-15 (for the three orbiter mission velocity levels). For liquid oxygen, autogenous pressurization was penalized because tight control over tank liquid/vapor heat and mass transfer processes must be exercised to prevent excessive vent losses and/or detrimental interaction with the screen propellant acquisition device. For this reason, cold gas helium pressurization was preferred for liquid oxygen storage.

For liquid hydrogen, however, autogenous pressurization possessed a large weight advantage and was favored over helium pressurization. Comparative ratings among APS/OMS integration options were more competitive than for the pressurization concepts; therefore selection of a preferred approach was more difficult. Separate nonrefillable tankage was preferred for both oxygen and hydrogen, primarily because of subsystem integration simplicity and fewer technological requirements associated with propellant acquisition (moderate screen sizes and no refill requirement).



APS	OMS (RL-10)
O/F = 3.0	O/F = 5.0
I_{SP} = 386 SEC	I_{SP} = 444 SEC
P_{TANK} = 38 (O_2)	P_{TANK} = 38 (O_2)
(LB_F/IN^2A) 48 (H_2)	(LBF/IN^2A) 29 (H_2)

APS/OMS PERFORMANCE

FIGURE C-9

	POINTS
TECHNOLOGY (20%)	
- PROPELLANT ACQUISITION	10
- REUSABLE INSULATION	5
- ON-ORBIT PROPELLANT TRANSFER (TANK REFILL)	3
- PRESSURIZATION	2
SIMPLICITY (15%)	
- NUMBER OF COMPONENTS	5
- OPERATIONAL COMPLEXITY	5
- SUBSYSTEM INTEGRATION COMPLEXITY	3
- CONTROL REQUIREMENTS	2
WEIGHT AND VOLUME (25%)	25
MISSION FLEXIBILITY (25%)	
- OPERATING CONSTRAINTS	10
- SENSITIVITY TO MAX ΔV INCREMENTS	5
- SENSITIVITY TO TOTAL IMPULSE	5
- SENSITIVITY TO THRUST	5
DEVELOPMENT (15%)	
- ENVIRONMENTAL SIMULATION	6
- INTEGRATED TEST REQUIREMENTS	6
- FACILITY REQUIREMENTS	3
TOTAL	100

EVALUATION CRITERIA

FIGURE C-10

CONCEPT	
TANKAGE	PRESS
FULLY INTEGRATED	COLD H_e
SEPARATE	COLD H_e
SEPARATE REFILLABLE	COLD H_e
COMBINED	COLD H_e
COMBINED REFILLABLE	COLD H_e
FULLY INTEGRATED	AUTOGENOUS
SEPARATE	AUTOGENOUS
SEPARATE REFILLABLE	AUTOGENOUS
COMBINED	AUTOGENOUS
COMBINED REFILLABLE	AUTOGENOUS

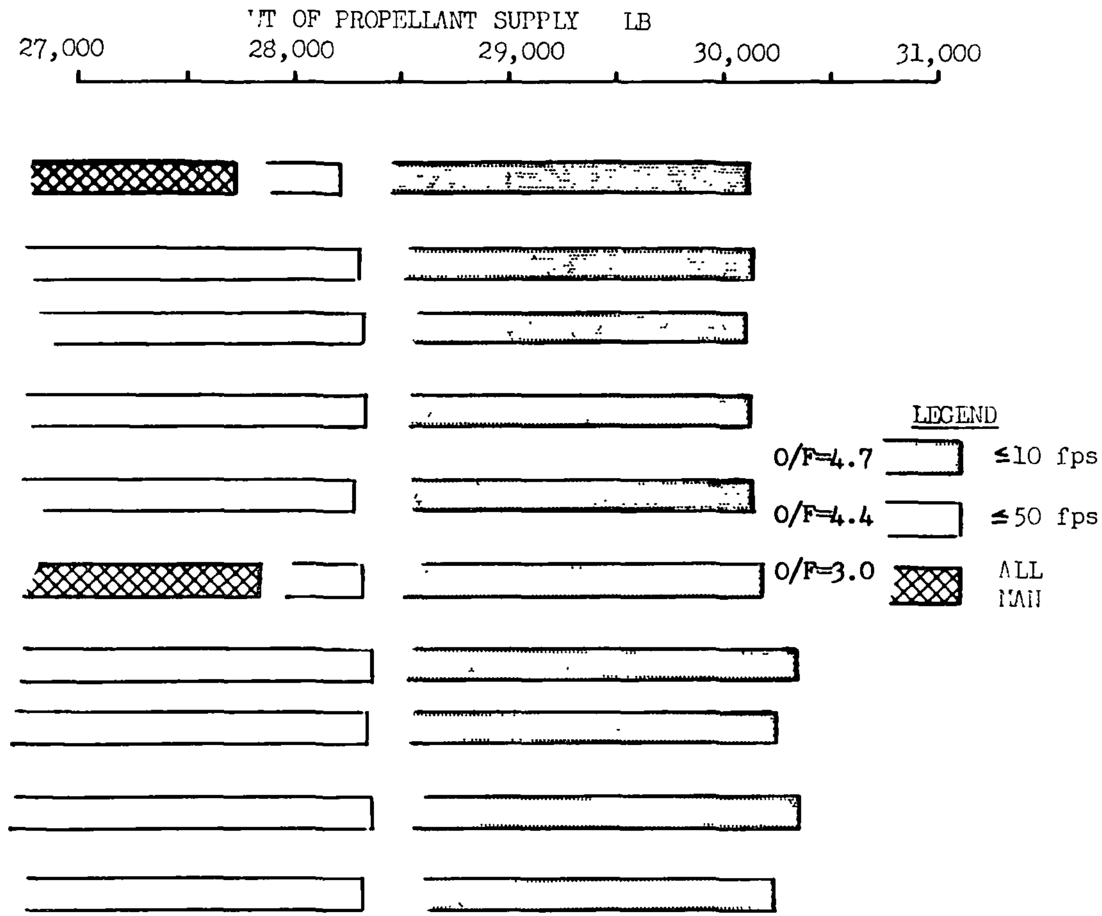
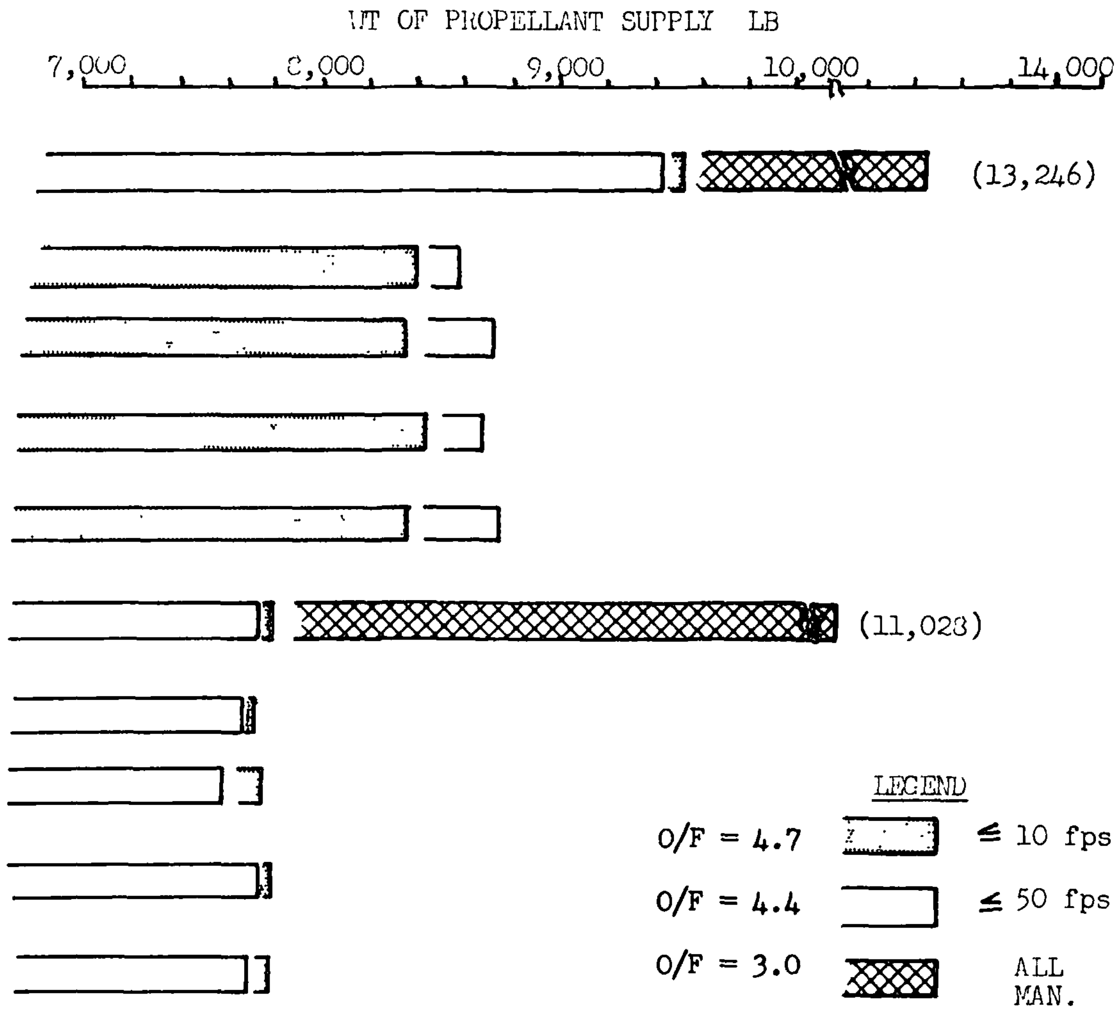


FIGURE C-11

LIQUID STORAGE
OXYGEN TANKAGE CONCEPTS

CONCEPT	
TANKAGE	PRESS
FULLY INTEGRATED	COLD H_e
SEPARATE	COLD H_e
SEPARATE REFILLABLE	COLD H_e
COMBINED	COLD H_e
COMBINED REFILLABLE	COLD H_e
FULLY INTEGRATED	AUTOGENOUS
SEPARATE	AUTOGENOUS
SEPARATE REFILLABLE	AUTOGENOUS
COMBINED	AUTOGENOUS
COMBINED REFILLABLE	AUTOGENOUS



LIQUID STORAGE
HYDROGEN TANKAGE CONCEPTS

FIGURE C-12

$\leq$ 10 fps

CONCEPT	SELECTION CRITERIA TECHNOLOGY (20%)	FLEXIBILITY (25%)	WEIGHT & VOL (25%)	DEVELOPMENT (15%)	SIMPLICITY (15%)	TOTAL
			O_2/H_2			
COLD He PRESSURIZATION						
FULLY INTEGRATED	12/7	10/10	25/1	6/2	15/15	68/35
SEPARATE	14/13	12/12	25/16	6/6	13/13	70/60 (✓)
SEPARATE REFILLABLE	9/9	16/16	25/16	6/6	8/8	64/55
COMBINED	14/13	11/11	25/15	6/6	10/10	66/55
COMBINED REFILLABLE	11/11	16/16	25/16	6/6	7/7	65/56
AUTOGENOUS PRESSURIZATION						
FULLY INTEGRATED	7/2	10/10	24/23	4/0	15/15	60/50
SEPARATE	9/8	12/12	23/25	4/4	13/13	61/62 (✓)
SEPARATE REFILLABLE	4/4	16/16	24/24	4/4	8/8	56/56
COMBINED	9/8	11/11	22/25	4/4	10/10	56/58
COMBINED REFILLABLE	6/6	16/16	24/23	4/4	7/7	57/56

(✓) BASELINE

FIGURE C-13

LIQUID STORAGE CONCEPT SELECTION

≤ 50 fps

CONCEPT	SELECTION CRITERIA: TECHNOLOGY (20%)	FLEXIBILITY (25%)	WEIGHT & VOL. (25%)	DEVELOPMENT (15%)	SIMPLICITY (15%)	TOTAL
			O_2/H_2			
COLD He PRESSURIZATION						
FULLY INTEGRATED	12/7	10/10	25/2	6/2	15/15	68/36
SEPARATE	14/11	12/12	24/12	6/6	13/13	69/54 (✔)
SEPARATE REFILLABLE	9/8	16/16	24/11	6/6	8/8	68/49
COMBINED	14/11	11/11	24/12	6/6	10/10	65/50
COMBINED REFILLABLE	11/10	16/16	24/10	6/6	7/7	64/49
AUTOGENOUS PRESSURIZATION						
FULLY INTEGRATED	7/2	10/10	24/23	4/0	15/15	60/50
SEPARATE	9/6	12/12	23/24	4/4	13/13	61/59 (✔)
SEPARATE REFILLABLE	4/3	16/16	24/25	4/4	8/8	56/56
COMBINED	9/6	11/11	23/23	4/4	10/10	51/54
COMBINED REFILLABLE	6/5	16/16	24/24	4/4	7/7	57/56

(✔) BASELINE

FIGURE C-14

LIQUID STORAGE CONCEPT SELECTION

ALL MANEUVERS

CONCEPT	SELECTION CRITERIA: TECHNOLOGY (20%)	FLEXIBILITY (25%)	WEIGHT&VOL (25%)	DEVELOPMENT (15%)	SIMPLICITY (15%)	TOTAL
			O_2/H_2			
COLD He PRESSURIZATION						
FULLY INTEGRATED	12/7	10/10	25/0	6/2	15/15	68/34 (✓)
SEPARATE						
SEPARATE REFILLABLE						
COMBINED						
COMBINED REFILLABLE						
AUTOGENOUS PRESSURIZATION						
FULLY INTEGRATED	7/2	10/10	24/25	4/0	15/15	60/52 (✓)
SEPARATE						
SEPARATE REFILLABLE						
COMBINED						
COMBINED REFILLABLE						

(✓) BASELINE

FIGURE C-15

LIQUID STORAGE CONCEPT SELECTION

Total APS/OMS tankage weights of supercritical propellant storage concepts are compared in Figure C-16 for both low and intermediate maneuver level missions. Because of reduced tank volume, refillable APS tankage concepts offered a significant weight advantage. Detailed ratings of supercritical storage concepts for both mission velocity levels are tabulated in Figure C-17. The nonrefillable concepts were rated superior in every category except weight. Thus, despite the large weight advantage associated with refillable tankage, the nonrefillable concept was preferred by a slight margin for the low maneuver level mission. However, at intermediate velocity levels, this weight advantage was overriding, and the refillable tankage concept was selected. Supercritical storage was not considered competitive with liquid storage for the high velocity level mission, due to excessive weight penalty. This is illustrated by Figure C-18, which compares liquid and supercritical APS storage weights for separate nonrefillable tanks, and shows the weight penalty would be in excess of 10K lb.

Based on these concept screening studies, both liquid and supercritical propellant storage were considered competitive for low and intermediate velocity level missions, whereas only liquid storage was considered attractive for the high velocity mission. These concepts, selected for further evaluation in Subtask A study, are illustrated in Figure C-19. The selected liquid storage concept employed separate, nonrefillable propellant tanks. Cold gas, helium pressurization was selected for liquid oxygen, and autogenous pressurization for liquid hydrogen. For the supercritical storage concept, separate nonrefillable tanks were employed for the low velocity mission, while separate refillable tanks were preferred for intermediate velocity levels.

C-2.2 Orbiter Propellant Thermal Conditioning - Preliminary evaluations of the three orbiter A mission duty cycles revealed that propellant supplied to the main engine tanks must be thermally conditioned in order to maintain satisfactory propellant feed pressures and temperatures. This is illustrated by orbiter A main engine tank pressure profiles for the largest +X maneuver in each of the three mission velocity levels (Figure C-20). As an example, for a velocity change of 50 ft/sec, the hydrogen supply must be superheated to approximately 200 R to maintain tank pressure above 20 lb/in^2a. Energy required for superheating could be supplied either actively (using a gas generator/heat exchanger assembly) or passively (using a heat sink type heat exchanger). These two approaches are illustrated in Figure C-21 and the analytical models used to compare approaches are described in the following paragraphs.

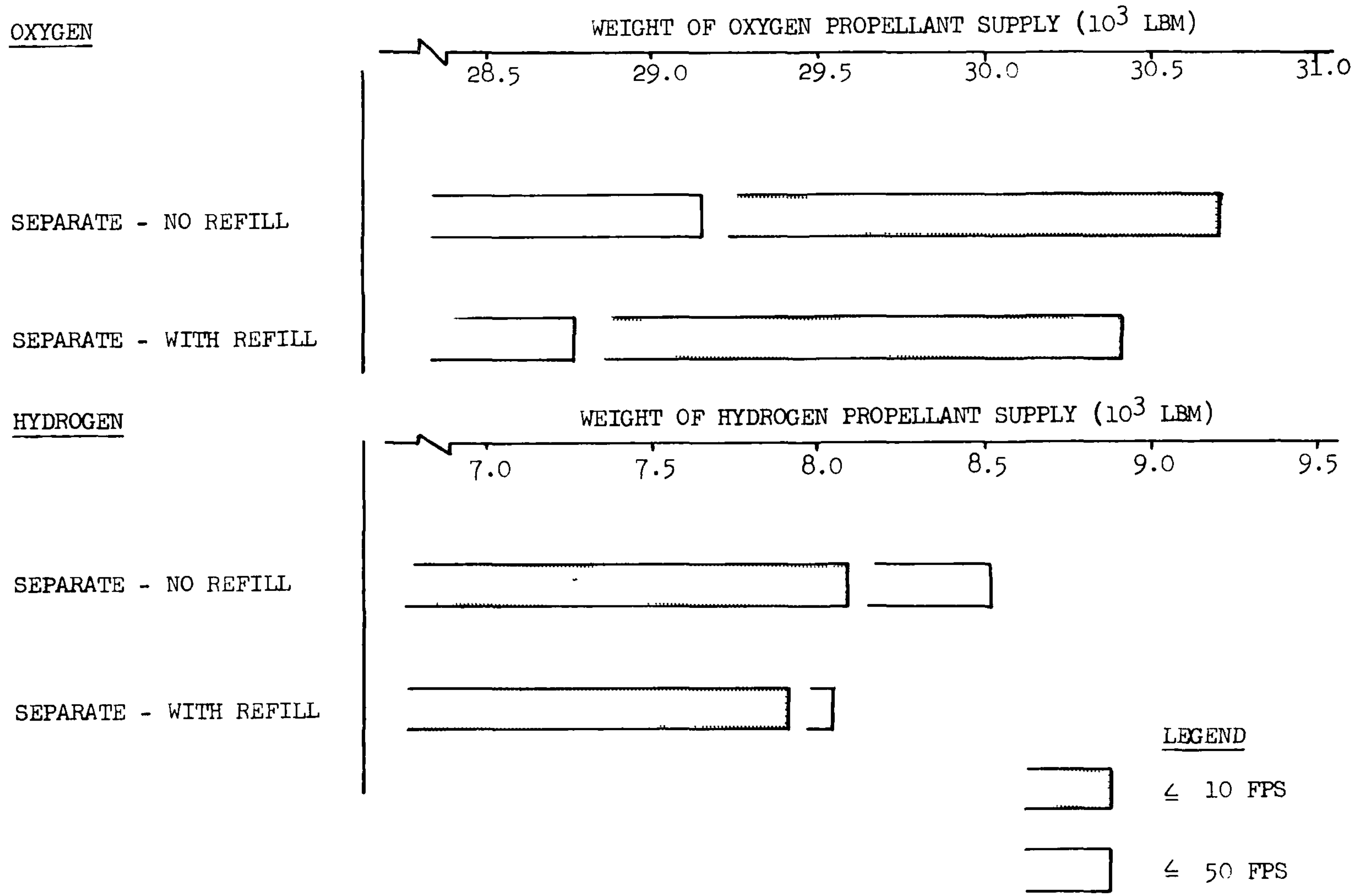


FIGURE C-16

SUPERCRITICAL TANKAGE CONCEPT COMPARISON

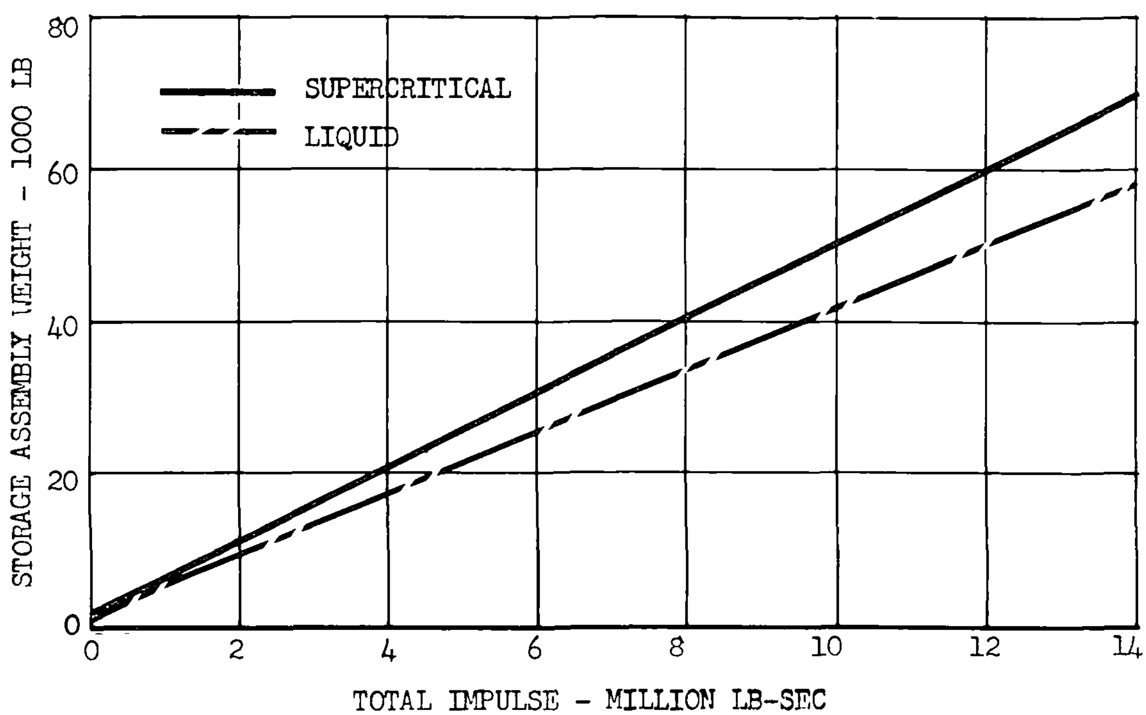
SELECTION CRITERIA	SUPERCRITICAL/PASSIVE			
	≤10 FT/SEC		≤50 FT/SEC	
	REFILLABLE	NON-REFILLABLE	REFILLABLE	NON-REFILLABLE
TECHNOLOGY (20%)				
- PROPELLANT ACQUISITION (10)	9	9	9	9
- INSULATION VENTING (5)	3	3	3	3
- TANK REFILL (3)	0	3	0	3
- PRESSURIZATION (2)	1	1	1	1
	(13)	(16)	(13)	(16)
SIMPLICITY (15%)				
- NO. OF COMPONENTS (5)	4	5	4	5
- OPERATIONAL COMPLEXITY (5)	3	4	3	4
- INTEGRATION COMPLEXITY (3)	2	2	2	2
- CONTROL REQUIREMENTS (2)	1	1	1	1
	(10)	(12)	(10)	(12)
WEIGHT AND VOLUME (25%)				
- WEIGHT (1 PER 80 LBS)	11	5	25	16
- VOLUME (1 PER 60 FT^3)	0	0	0	0
	(11)	(5)	(25)	(16)
MISSION FLEXIBILITY (25%)				
- OPERATING CONSTRAINTS (10)	6	8	6	8
- SENSITIVITY TO MAX ΔV INCREMENTS (5)	3	3	3	3
- SENSITIVITY TO TOTAL IMPULSE (5)	4	3	4	3
- SENSITIVITY TO THRUST LEVEL (5)	2	2	2	2
	(15)	(16)	(15)	(16)
DEVELOPMENT (15%)				
- ENVIRONMENTAL SIMULATION (6)	2	3	2	3
- INTEGRATED TEST REQUIREMENTS (6)	2	3	2	3
- FACILITY AVAILABILITY (3)	2	1	2	1
	(6)	(7)	(6)	(7)
TOTAL	(55)	(56)√	(69)√	(67)

√ SELECTED

FIGURE C-17

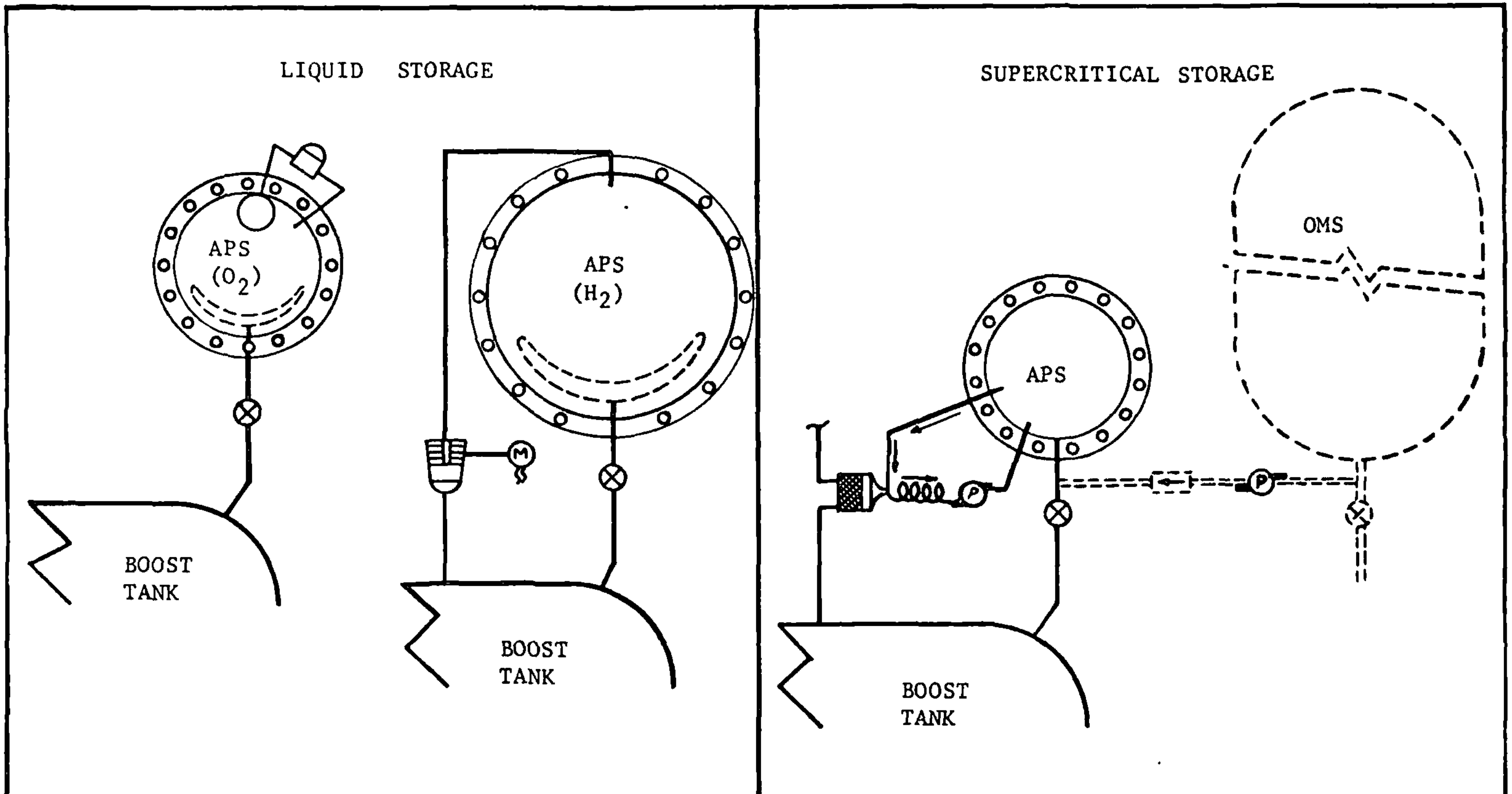
SUPERCRITICAL STORAGE CONCEPT SELECTION

- SEPARATE NON-REFILLABLE TANKAGE
- H_2 AND O_2 TANKS



APS PROPELLANT STORAGE ASSEMBLY WEIGHT SENSITIVITY

FIGURE C-18



LIQUID STORAGE — APPLICABLE ORBITER MISSIONS

- +X AXIS MANEUVERS $\leq$ 10 FT/SEC
- +X AXIS MANEUVERS $\leq$ 50 FT/SEC
- ALL MANEUVERS

SUPERCRITICAL STORAGE — APPLICABLE ORBITER MISSIONS

- +X AXIS MANEUVERS $\leq$ 10 FT/SEC (NON-REFILLABLE)
- +X AXIS MANEUVERS $\leq$ 50 FT/SEC (REFILLABLE)

FIGURE C-19

SELECTED TANKAGE CONCEPTS

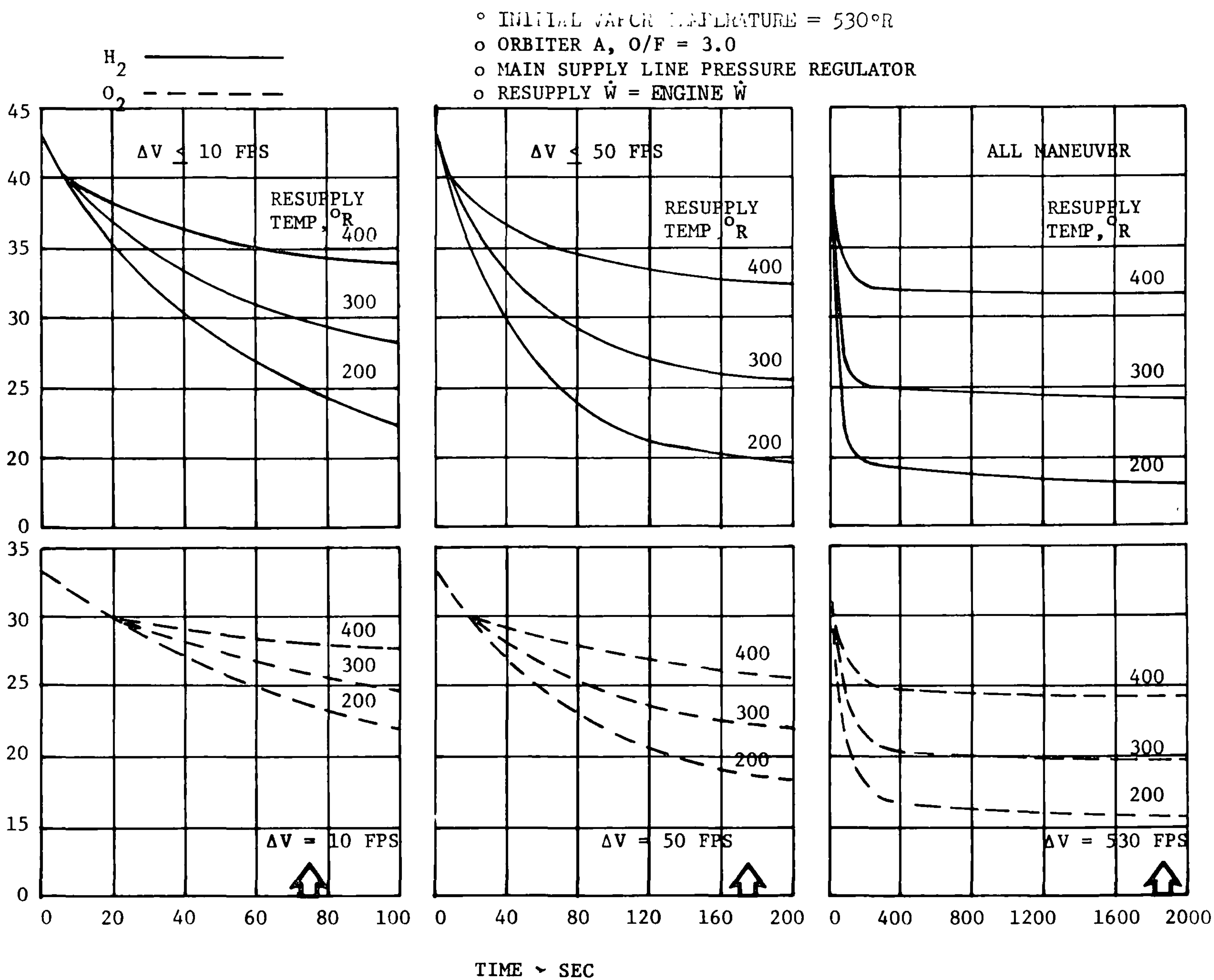
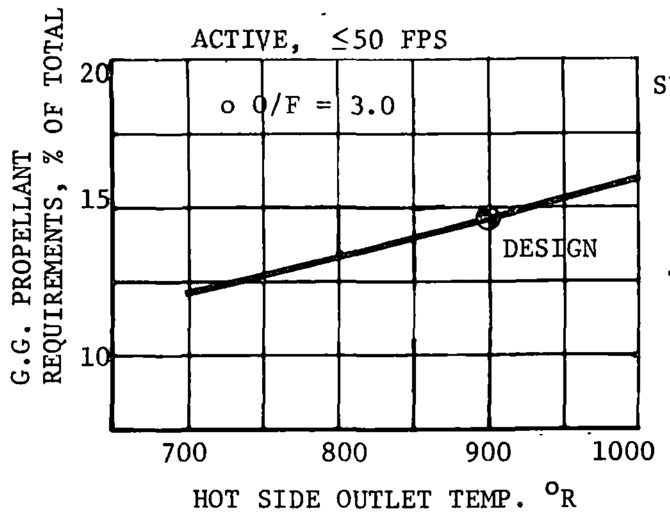
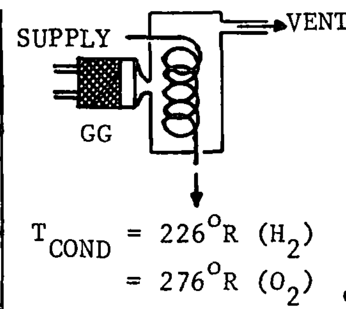


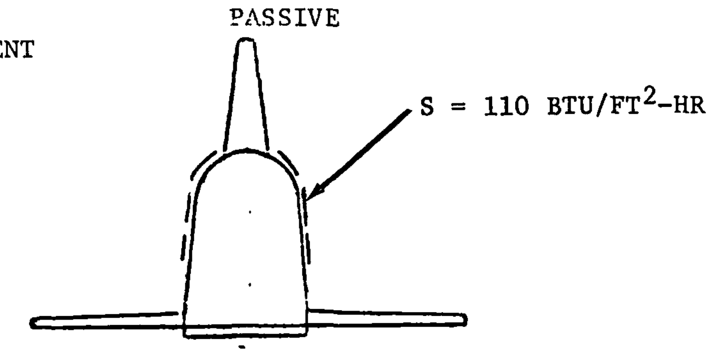
FIGURE C-20

SENSITIVITY OF BOOST TANK PRESSURE TO CONDITIONING TEMPERATURE

ORBITER A







- HEAT EXCHANGERS ON EACH SIDE OF VEHICLE
- CONTROLS SELECT HOT SIDE OR ALLOW FLOW THROUGH BOTH SIDES DURING LOW HEATING PERIODS

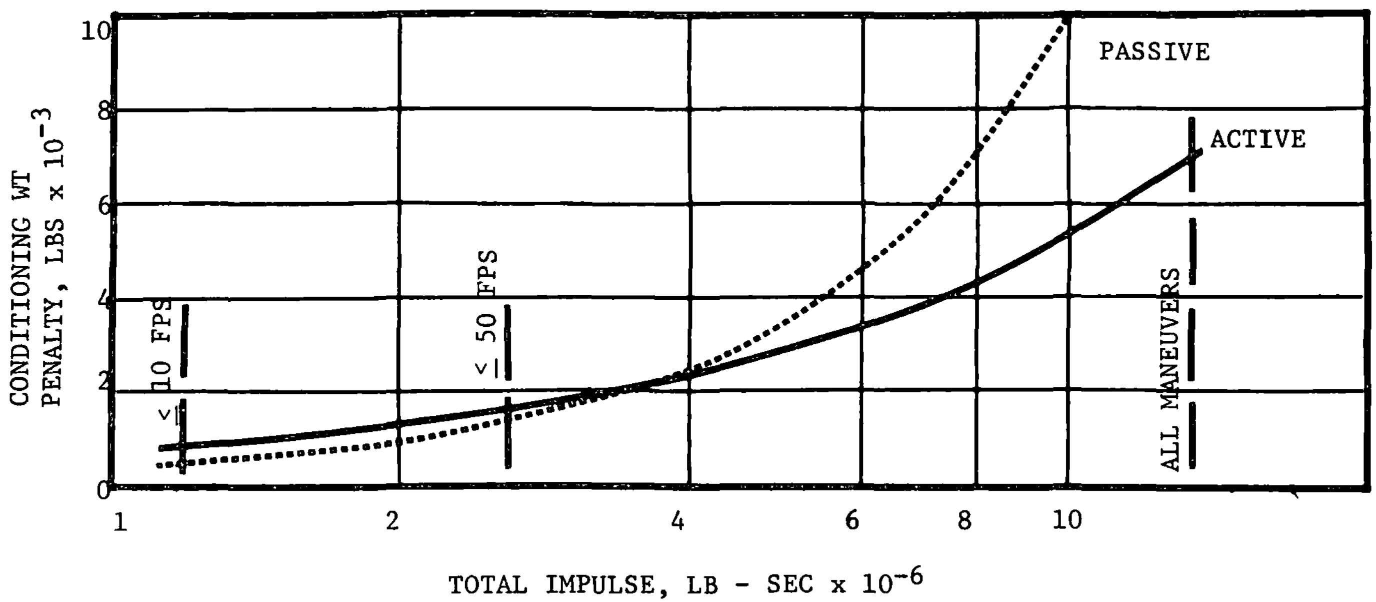


FIGURE C-21

PASSIVE/ACTIVE HEAT EXCHANGER COMPARISON

The active heat exchanger design was a helical tube and shell configuration, using bipropellant gas generator exhaust products as the thermal energy source. Gas generator temperature was limited to 2000 R to promote long service life. To ensure that water vapor contained in the exhaust products would not freeze in the vent line, the heat exchanger hot side outlet temperature was limited to 900 R. Therefore, energy available for superheating main engine tank resupply propellant was the enthalpy difference between gas generator products entering the heat exchanger at 2000 R, and leaving the heat exchanger at 900 R. Applying this enthalpy difference, the gas generator propellant requirement for conditioning main engine tank resupply (for the example of Figure C-21) was approximately 15 percent of total APS engine flow. Active heat exchanger/gas generator weights were computed using parametric data of Appendix D.

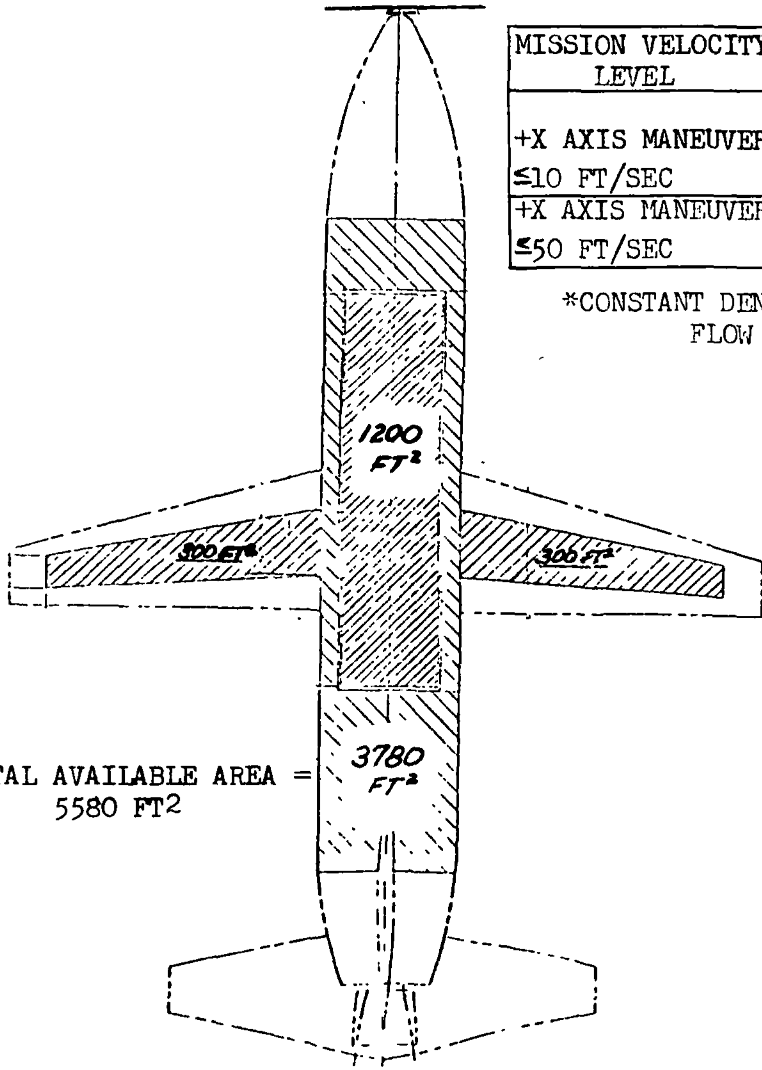
The passive heat exchanger concept selected for Subtask A study employed a number of straight tubes mounted to the inside surface of the vehicle skin (Figure C-21). Thus, the energy required for heating main engine tank resupply propellant was derived from the internal energy (mass and heat capacity) of both the vehicle skin and heat exchanger tubing and the radiant solar energy flux incident on the vehicle external surface. Heat exchanger surface area requirements were calculated to determine their compatibility with available vehicle surface area. The primary assumptions made for these calculations were:

(1) incident radiant solar energy flux equaled 110 Btu/hr-ft^2 (which is a conservative average value for low earth orbit)

(2) vehicle skin and heat exchanger tubing temperatures recovered fully between APS maneuver burns.

The calculated heat exchanger surface areas for both orbiters A and B are compared with available vehicle areas in Figure C-22. The required number of heat exchanger tubes was determined based on a tube diameter of 0.325 in and a tube spacing of 2.0 in. This diameter provided an exit Mach number of 0.3 at the required exit propellant temperature. Passive heat exchanger tubing weights were then estimated assuming aluminum tubing having a minimum gage thickness of 0.022 in.

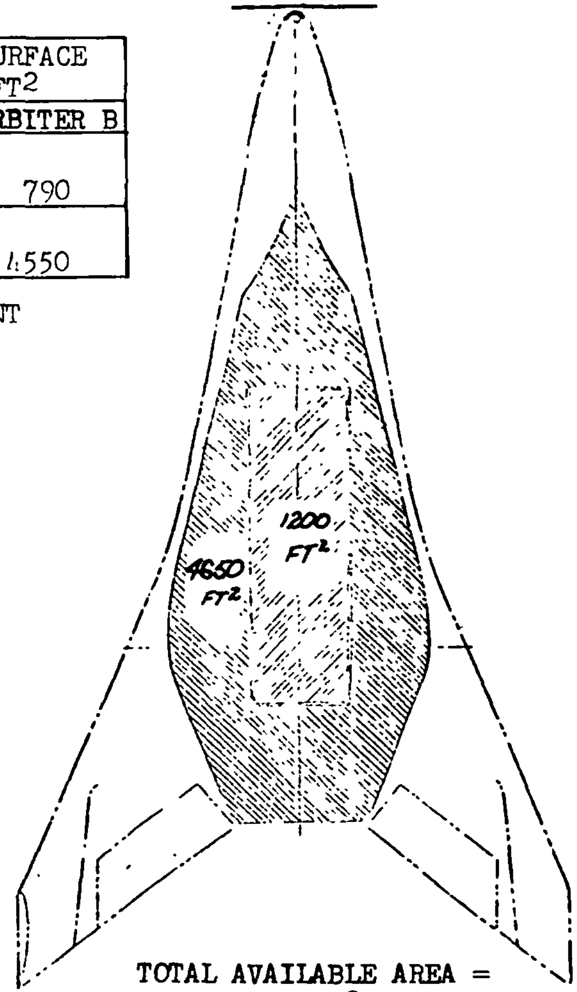
Curves for conditioning assembly weight presented in Figure C-21 compare active conditioning penalty (gas generator propellant requirement plus heat exchanger and gas generator weight) with passive heat exchanger tubing weight.



TOTAL AVAILABLE AREA = 5580 FT^2

MISSION VELOCITY LEVEL	REQUIRED SURFACE AREAS * FT^2	
	ORBITER A	ORBITER B
+X AXIS MANEUVERS ≤10 FT/SEC	430	790
+X AXIS MANEUVERS ≤50 FT/SEC	2930	4550

*CONSTANT DENSITY PROPELLANT FLOW CONTROL



TOTAL AVAILABLE AREA = 5850 FT^2

AVAILABLE AREA

FOR

SURFACE HEAT EXCHANGERS

FIGURE C-22

As shown, both active and passive concepts were weight competitive for low ($\Delta V \leq 10$ ft/sec) and intermediate ($\Delta V \leq 50$ ft/sec) mission velocity levels. For the all- maneuver APS mission, however, passive conditioning weight was prohibitive. Weight penalty for the active approach can be reduced if a propulsive vent of the gas generator exhaust products is employed during +X axis maneuver burns. From the example given in Figure C-23, it is seen that gas generator propellant requirements can be reduced by approximately 2, 11 and 19 percent for low, intermediate and high mission velocity levels. Based on these comparative weights, both active and passive thermal conditioning concepts were selected for further investigation at low and intermeditate velocity levels during Subtask A. However, due to the passive concept's excessive weight penalty only active thermal conditioning was considered practical for the high velocity level mission.

C.2.3 Propellant Flow Control - Results of parametric APS engine performance studies (Figures C-24 and C-25) showed significant thrust and mixture ratio sensitivity to engine inlet pressure and temperature. For the purpose of preventing excessive thrust and mixture ratio excursions during orbiter and booster APS missions, alternate propellant flow control schemes were investigated.

Orbiter - During Subtask A study, four primary flow control concepts evolved for the orbiter, each providing progressively more accurate control of engine thrust and mixture ratio. These include:

(1) engine tank mass addition
(2) main engine tank mass addition with differential pressure regulators in individual engine feedlines.
(3) mass addition with main supply line pressure regulator, and
(4) mass addition with both a main supply line pressure regulator and a liquid/vapor mixing chamber.

Illustrations of these four control concepts are presented in Figure C-26. In the mass addition approach, main engine tank pressure was allowed to decay to a prescribed level, at which time thermally conditioned resupply propellant was injected into the tanks. With resupply only, loose control was exercised over main engine tank pressure and temperature. Typical operating characteristics of this approach are shown in Figure C-27 for an orbiter A velocity change of 50 ft/sec. As shown, engine thrust and mixture ratio variation were substantial.

In the second control concept, oxygen and hydrogen pressure at the engine

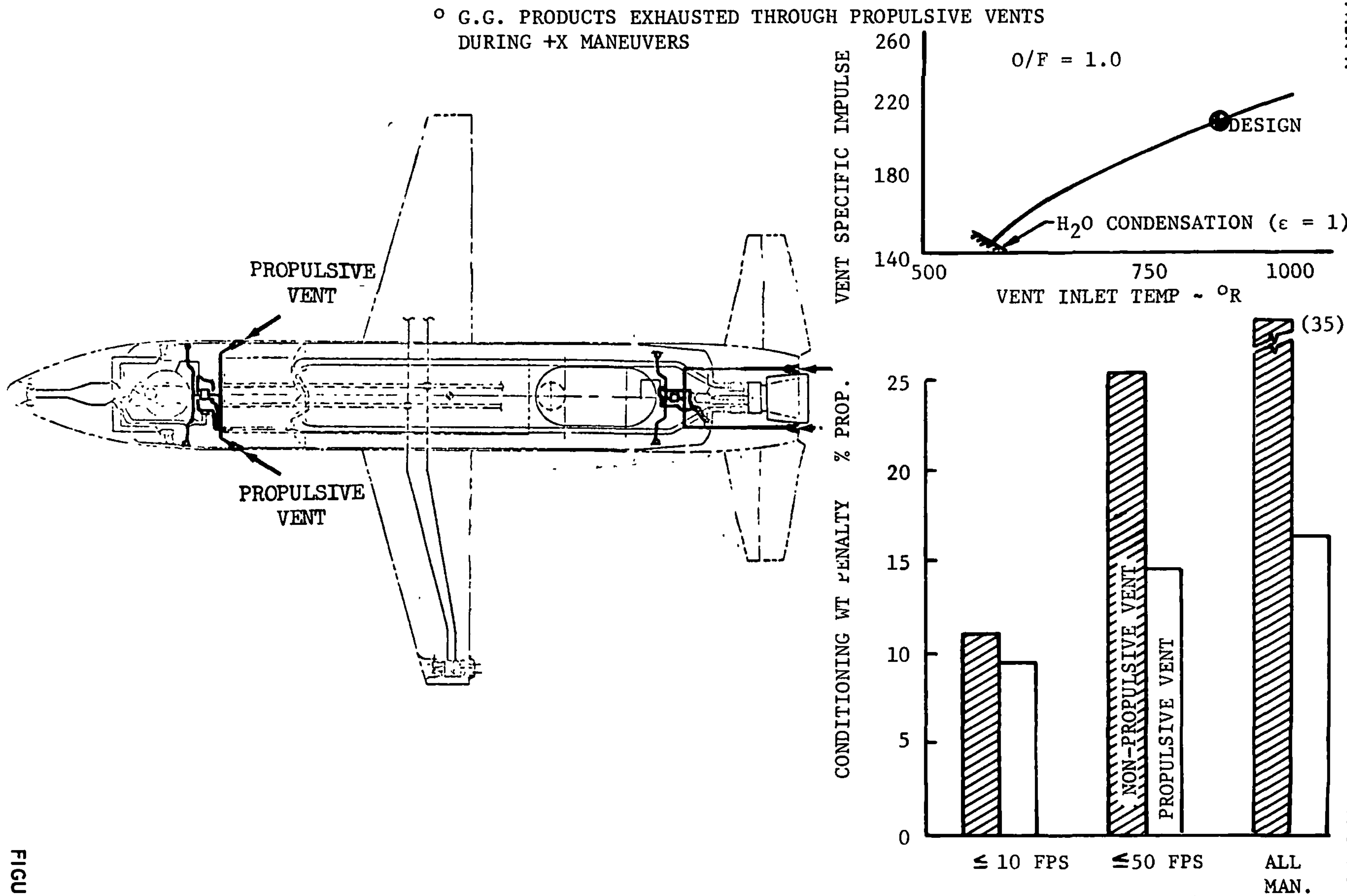


FIGURE C-23

PROPULSIVE VENT FOR ACTIVE HEAT EXCHANGER

DESIGN POINT

F = 1080 LBS
MR = 3.0 $\epsilon = 8.1$
$P_C = 13.7$
$P_{INLET} = 15.7$
$T_{INLET} = 150°R\ (H_2)$
$200°R\ (O_2)$

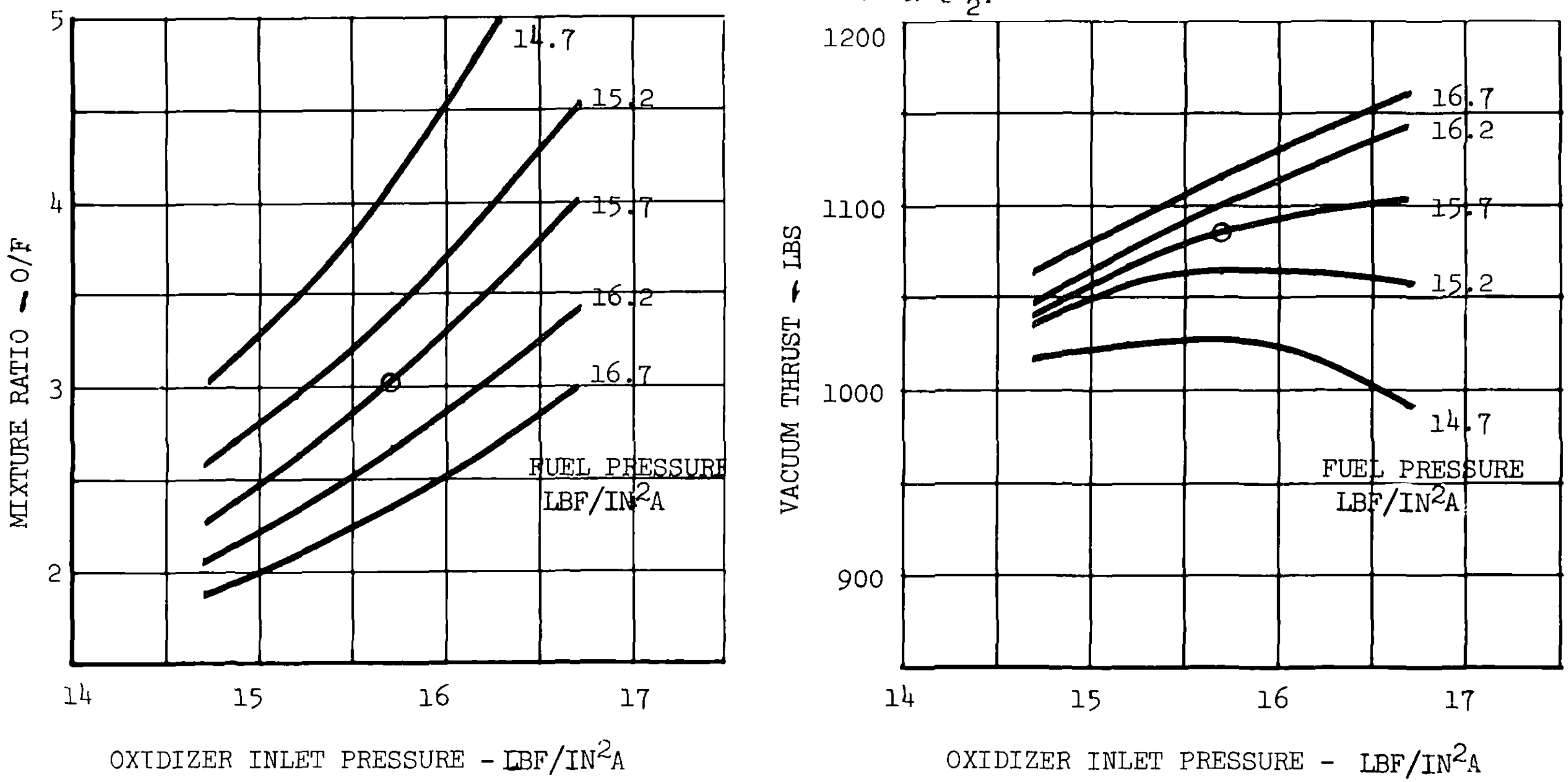
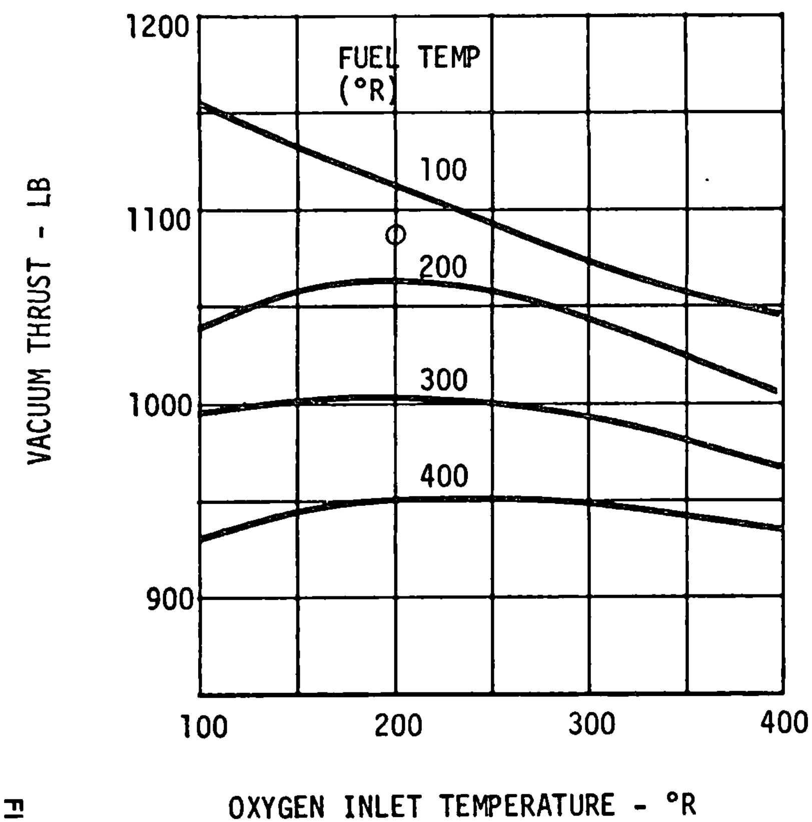


FIGURE C-24

ENGINE PERFORMANCE SENSITIVITY TO INLET PRESSURE

DESIGN POINT

P_c = 13.7 LBF/IN^2A
F = 1080 LBS
MR = 3.0
ϵ = 8:1
T INLET = 150°R (H_2)
200°R (O_2)



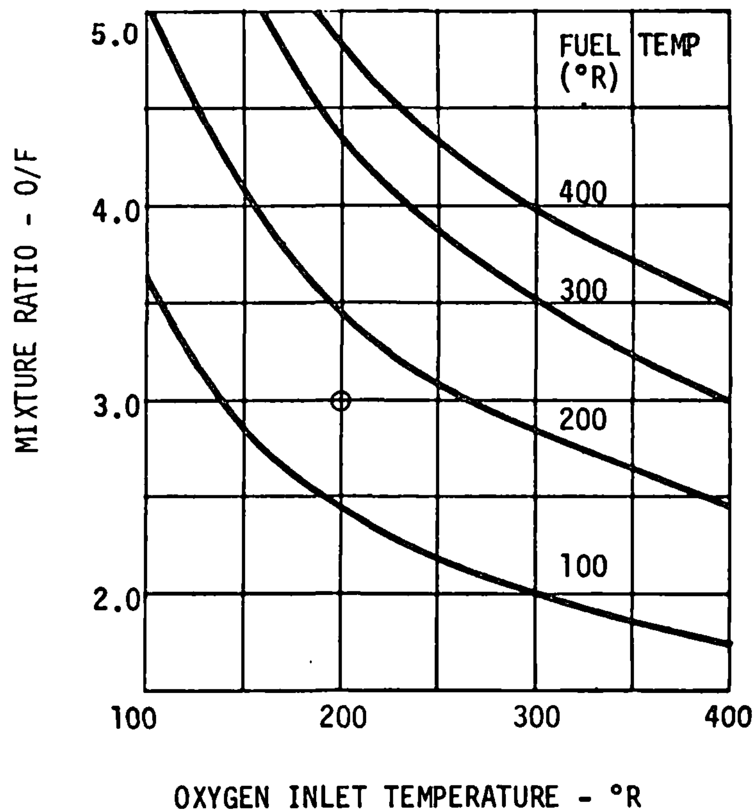


FIGURE C-25

ENGINE PERFORMANCE SENSITIVITY TO INLET TEMPERATURE

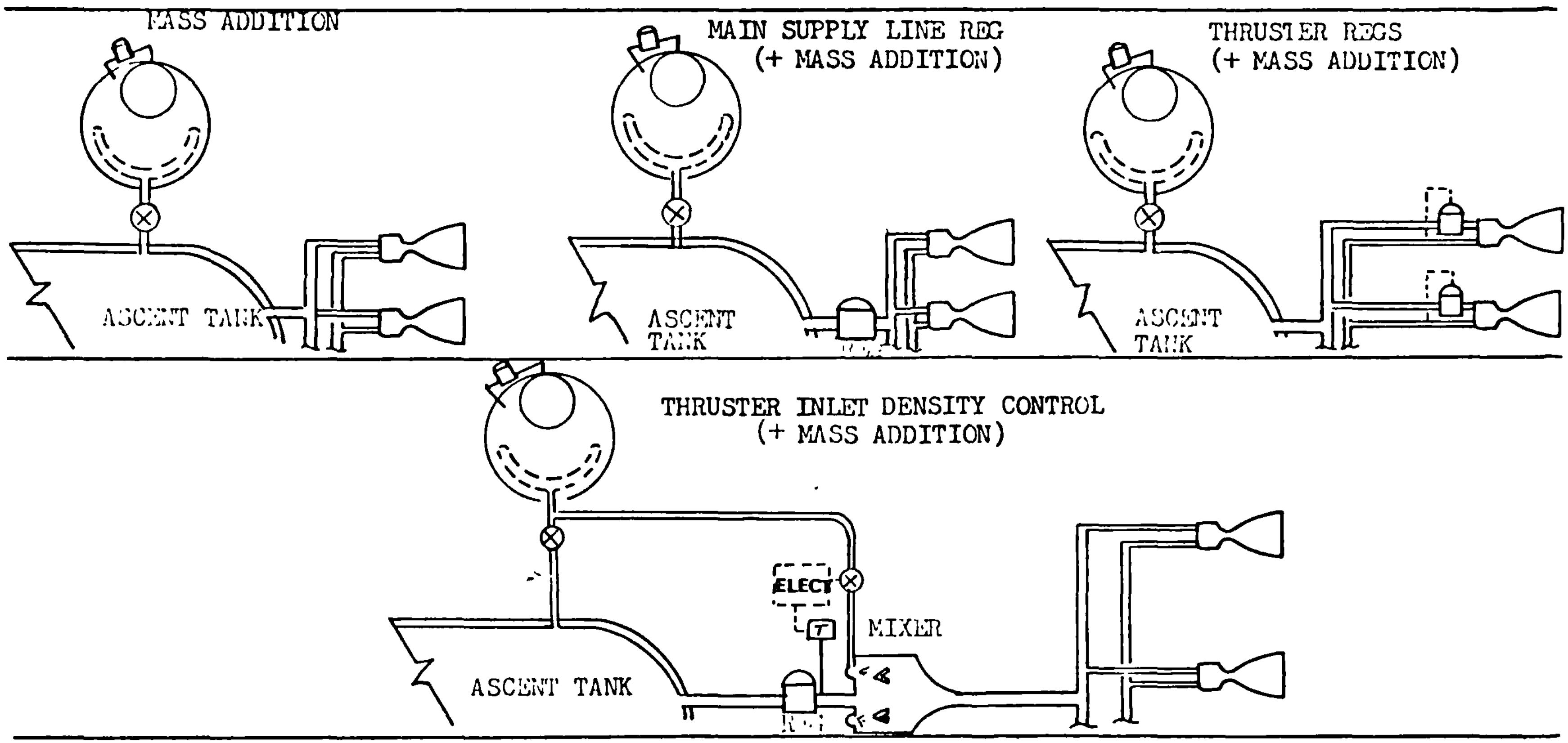
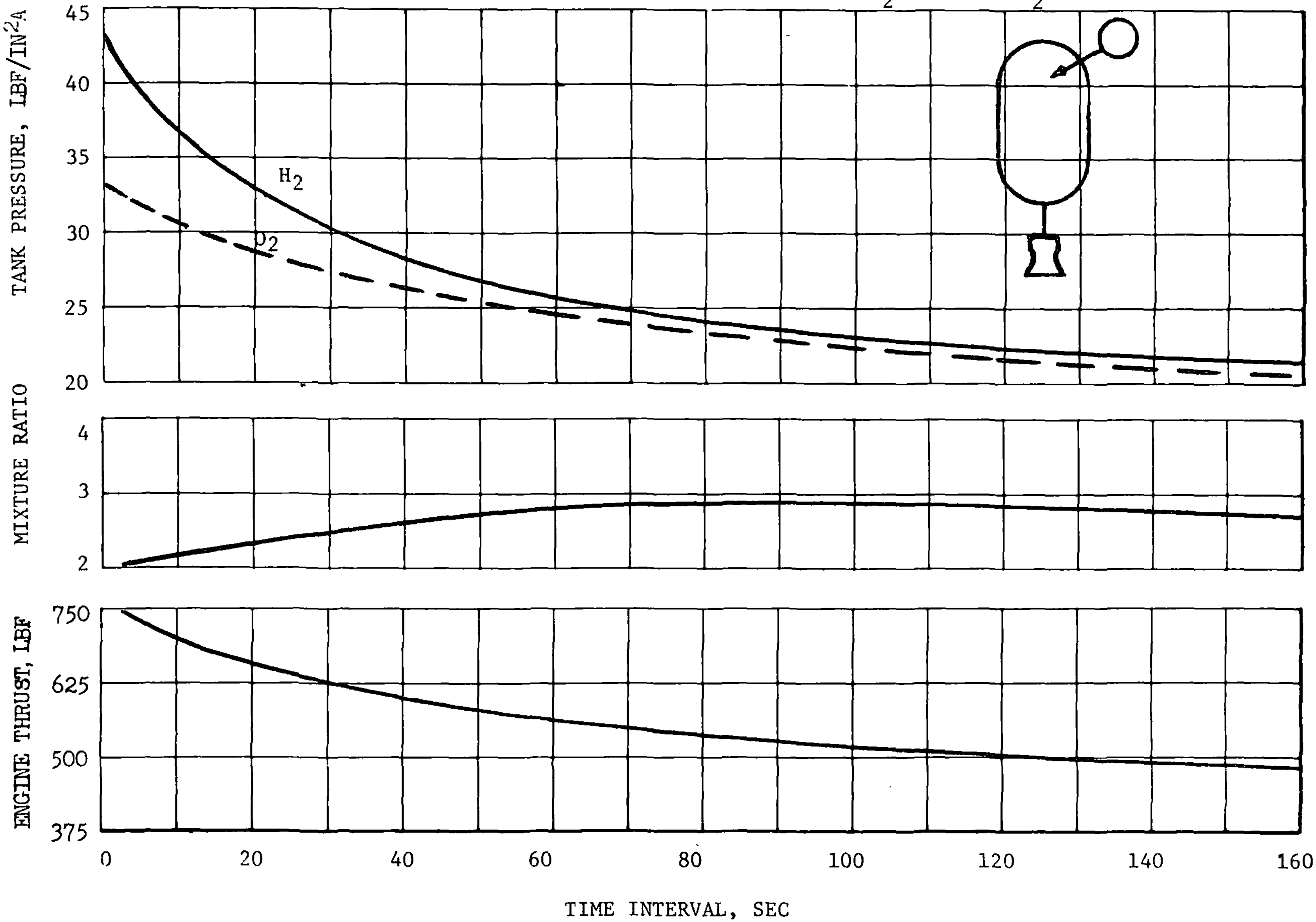


FIGURE C-26

ORBITER PROPELLANT FLOW CONTROL CONCEPTS

- O/F = 3.0
- VEHICLE A ORBITER (ΔV = 50 FPS)
- RATIO OF RESUPPLY TO ENGINE FLOW = 1
- INITIAL VAPOR TEMPERATURE = 530°R
- RESUPPLY TEMPERATURES : H_2 = 226°R, O_2 = 276°R



MASS ADDITION

FIGURE C-27

inlet were allowed to vary, but were maintained to a constant differential pressure by regulators on each engine. Operating characteristics for this differential regulator concept are presented in Figure C-28. As shown, relatively constant engine mixture ratio was achieved; however, engine thrust varied over a wide range since no control other than mass addition was exercised over main supply line propellant pressure (flow rate).

In the third concept, a main supply line pressure regulator provided inlet pressure control. As shown in Figure C-29, this provided relatively tight control of both thrust and mixture ratio. Slight variations which did result were caused by propellant temperature fluctuations as propellant was withdrawn from main engine tanks.

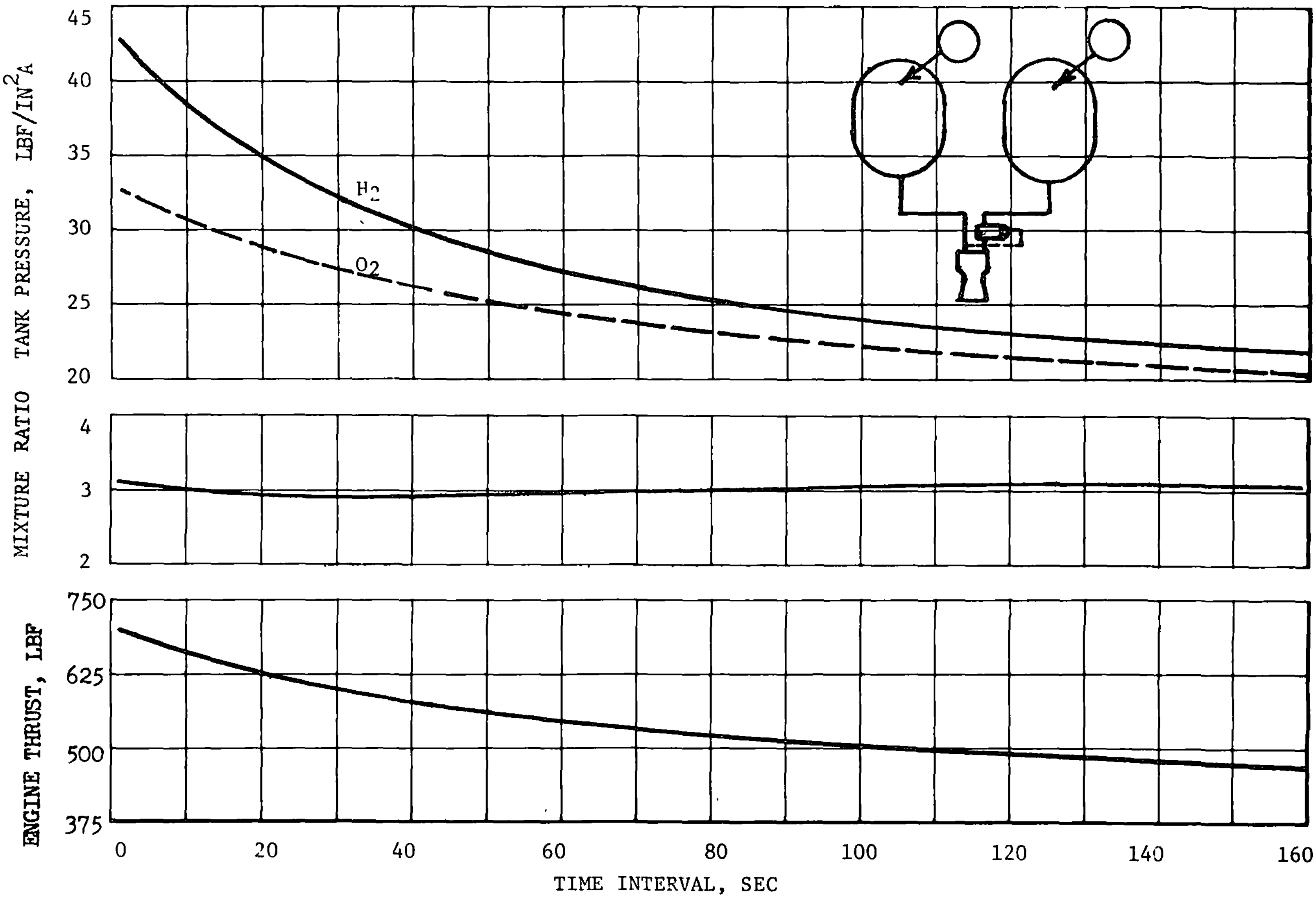
Most precise control of engine thrust and mixture ratio was achieved with the fourth approach by incorporating a propellant mixing chamber adjacent to the main supply line regulator. Typical operating characteristics for this last approach are shown in Figure C-30. During operation, gaseous propellant was withdrawn from main engine tanks, regulated to required feed pressure, and mixed with liquid propellant supplied directly from APS storage tanks. By throttling liquid flowrate, constant mixer outlet temperature was maintained. This was the most complex of the control concepts considered but energy requirements for the thermal conditioning assembly were minimized, since tank outflow and resupply rates were lower conpared with other control concepts. Therefore, heat exchanger length and tubing diameter for both active and passive conditioning concepts were reduced, and gas generator propellant demands were reduced for the active concept.

Each of these four control concepts were considered viable orbiter candidates and were investigated in greater depth during subsequent Subtask A studies. This further study, described in the body of this report, identified the most attractive concept in terms of weight, mission flexibility, simplicity, and technology development requirements.

Booster - Mission duty cycle simulations for shuttle boosters revealed that total impulse requirements could be satisfied entirely using residual gaseous propellant contained in main engine tanks following boost. Since this eliminated both propellant storage and thermal conditioning assemblies, neither mass addition nor liquid/vapor mixing flow control approaches were considered for the booster. As a result, only three booster concepts were investigated:

(1) a simple blowdown approach, exercising no control over propellant

- O/F = 3.0
- VEHICLE A ORBITER (ΔV = 50 FT/SEC)
- RATIO OF RESUPPLY TO ENGINE FLOW = 1
- INITIAL VAPOR TEMPERATURE = 530°R
- RESUPPLY TEMPERATURE: H_2 = 226°R, O_2 = 276°R



DIFFERENTIAL PRESSURE REGULATOR

FIGURE C-28

- O/F = 3.0
- VEHICLE A ORBITER (ΔV = 50 FPS)
- RATIO OR RESUPPLY TO ENGINE FLOW = 1
- INITIAL VAPOR TEMPERATURE = 530°R
- SUPPLY TEMPERATURES: H_2 = 226°R, O_2 = 276°R

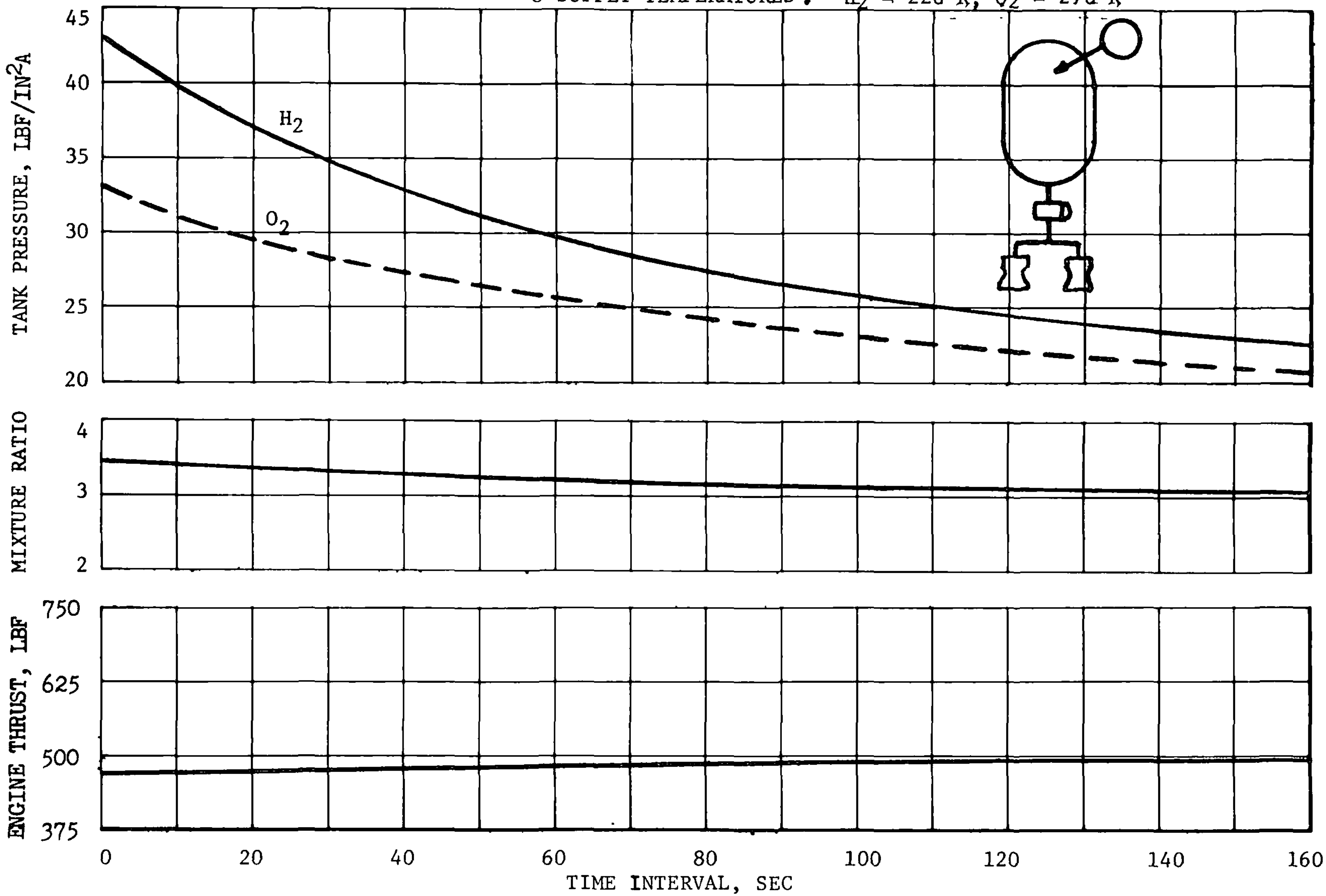


FIGURE C-29

MAIN SUPPLY LINE PRESSURE REGULATION

- O/F = 3.0
- VEHICLE A ORBITER (ΔV = 50 FT/SEC)
- RATIO OF RESUPPLY TO TANK OUTFLOW = 1
- INITIAL VAPOR TEMPERATURE = 530°R
- RESUPPLY TEMPERATURES: H_2 = 100°R, O_2 = 265°R

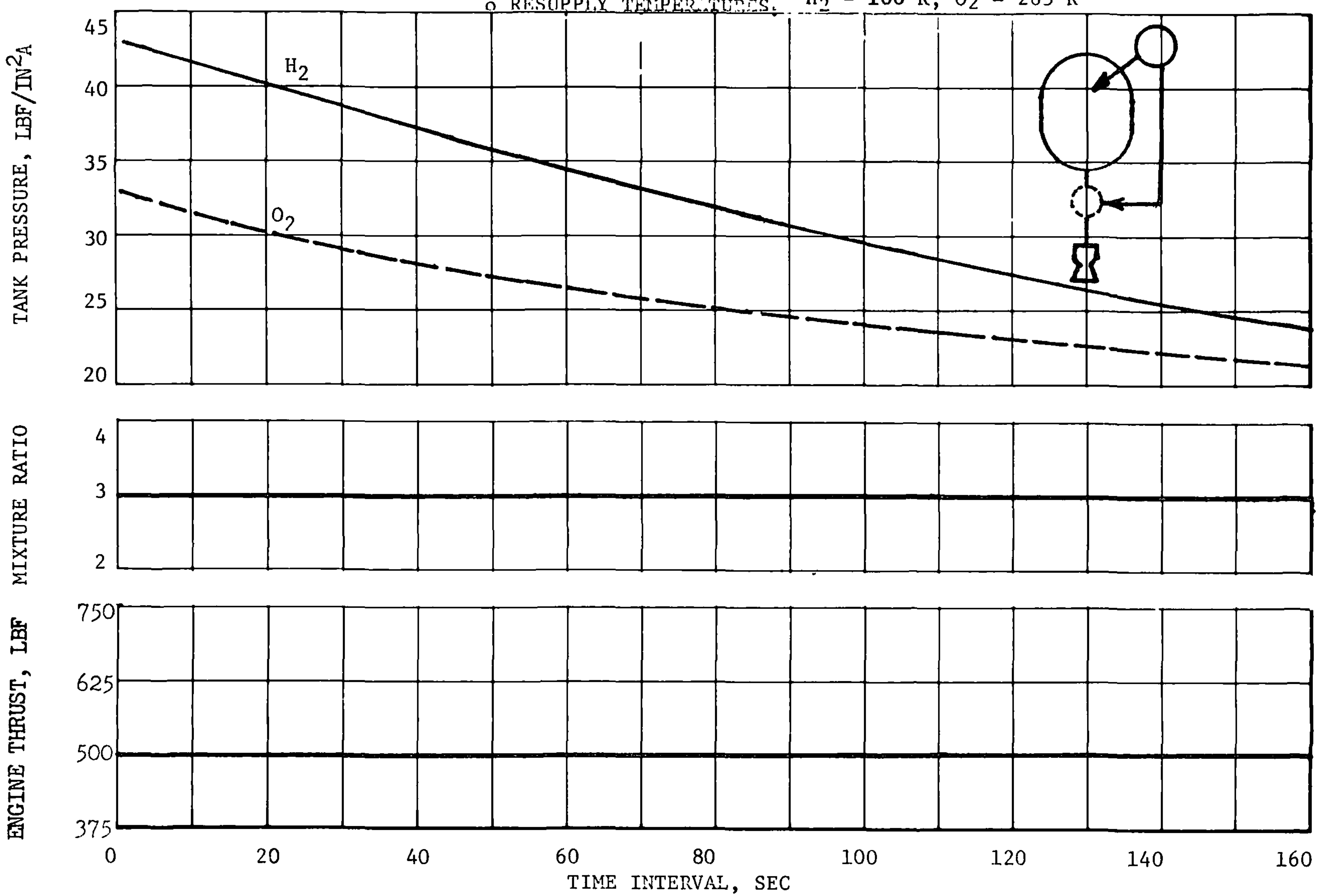


FIGURE C-30

CONSTANT DENSITY CONTROL

flow to the engine.

(2) a pressure regulated approach employing differential pressure regulators on hydrogen inlet lines to each engine assembly, and

(3) a pressure regulated approach employing main supply line regulators.

Operating characteristics of the three concepts are compared in Figure C-31 for booster A. The most interesting results obtained from this comparison were the nearly identical tank pressure/thrust profiles and the constancy of engine mixture ratio for all concepts. Although engine thrust decayed for blowdown and differential regulator concepts, minimum engine thrust levels delivered at end of the mission were sufficiently high to satisfy booster angular acceleration constraints with two engines out. Because of this result and the large line diameters which render very large main supply line regulators, only the first two booster flow control concepts were investigated further during Subtask A.

- INITIAL TANK VAPOR TEMPERATURES: H_2 = 200°R, O_2 = 300°R
- NO PROPELLANT RESUPPLY

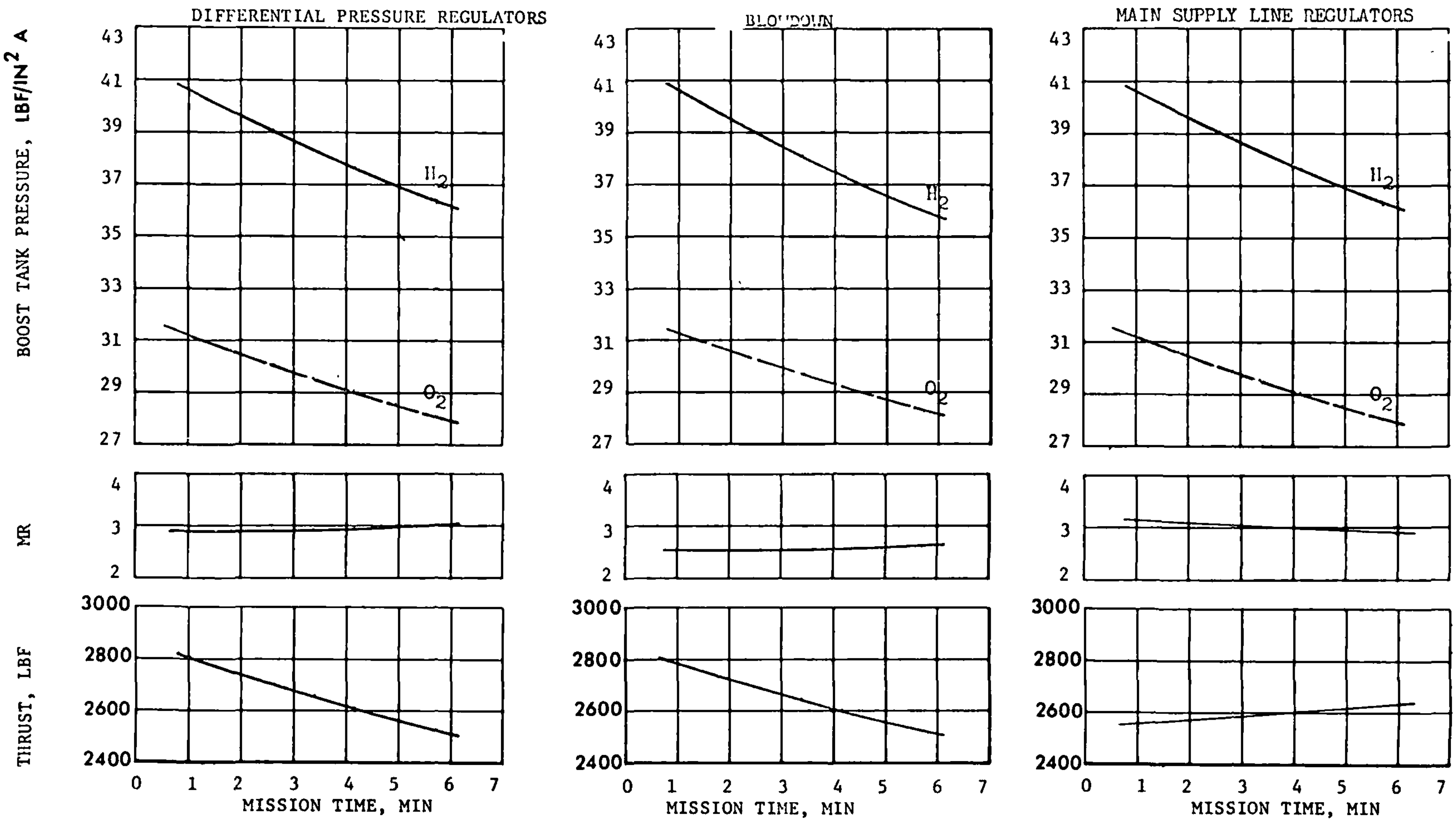


FIGURE C-31

VEHICLE A BOOSTER APS OPERATIONAL CHARACTERISTICS

APPENDIX D

Engine Assembly and Component Models

D-1. INTRODUCTION

Prior to performing synthesis and optimization of candidate APS concepts, analyses of APS engine assemblies and primary subsystem components were conducted to select reference component concepts and establish physical and performance characteristics. The components considered were:

(1) engine assembly
(2) active heat exchanger
(3) gas generator
(4) pressure regulator
(5) propellant transfer pump, and
(6) propellant shutoff valve

A discussion of physical and performance characteristics of propellant storage tank, passive heat exchanger, and main engine tanks is presented in Appendix C.

Realistic models for the above components which define weight, size, and performance as a function of design parameters were developed and employed in a sybsystem design and sizing computer program for evaluating subsystem concepts. A listing of ranges for which primary design parameters were developed is presented in Figure D-1. Presented in this appendix are a description of the selected design concepts, and a summary of physical and performance characteristics for each of the above components.

SYMBOL	DESCRIPTION	RANGE
F	Thrust Level	300-4,000 lbs.
P_c	Thruster Chamber Pressure	10-30 LBF/IN^{2}A
$(MR)_E$	Thruster Mixture Ratio	2.0-7.0 lbs O_2/lbs H_2
$(T_p)_E$	Thruster Propellant Inlet Temperature	200-600 (H_2)
		300-600 (O_2)
ϵ	Expansion Ratio	2 to 15
$\dot{W}$	Propellant Flow Rate	0.15-10.0 lbs/sec (H_2)
		0.60-45.0 lbs/sec (O_2)
		(1-6 equiv. thrusters)
T_{GG}	Gas Generator Temperature	1600-2500°R
P_{GG}	Gas Generator Pressure	10-30 LBF/IN^{2}A
$(MR)_{GG}$	Gas Generator Mixture Ratio	(consistant with T_{GG})
$(T_p)_{GG}$	Gas Generator Inlet Propellant	200-600 (H_2)
	Temperature	300-600 (O_2)
$(\Delta P)_H$	Heat Exchanger Pressure Drop	5-25 LBF/IN^{2}A
$(\Delta T)_{HOT}$	Heat Exchanger Hot Side Temperature Drop	600-2000°R

LOW PRESSURE APS

PARAMETER DEFINITION

FIGURE D-1

D-2. ENGINE ASSEMBLY

The selected low pressure APS engine concept is shown in Figure D-2. The engine consists of injector assembly, fuel film-cooled thrust chamber, electrical spark igniter assembly, and propellant control valves. The injector is a coaxial element concept sized for gaseous propellants. The rolled and formed combustion chamber is fabricated of Hastelloy steel. The fuel film coolant required to cool the chamber was determined by thermal analysis with consideration of thermal cycle and creep stress rupture. An electric spark igniter was selected based on its quick response, long life capability, and demonstrated engine ignition capability. The igniter design utilized separate control valves to provide positive propellant control to the sequenced torch igniter chamber. Diaphragm actuated poppet valves, shown in Figure D-3, were selected because they were light weight and offered potential fast response and high cycle life capability.

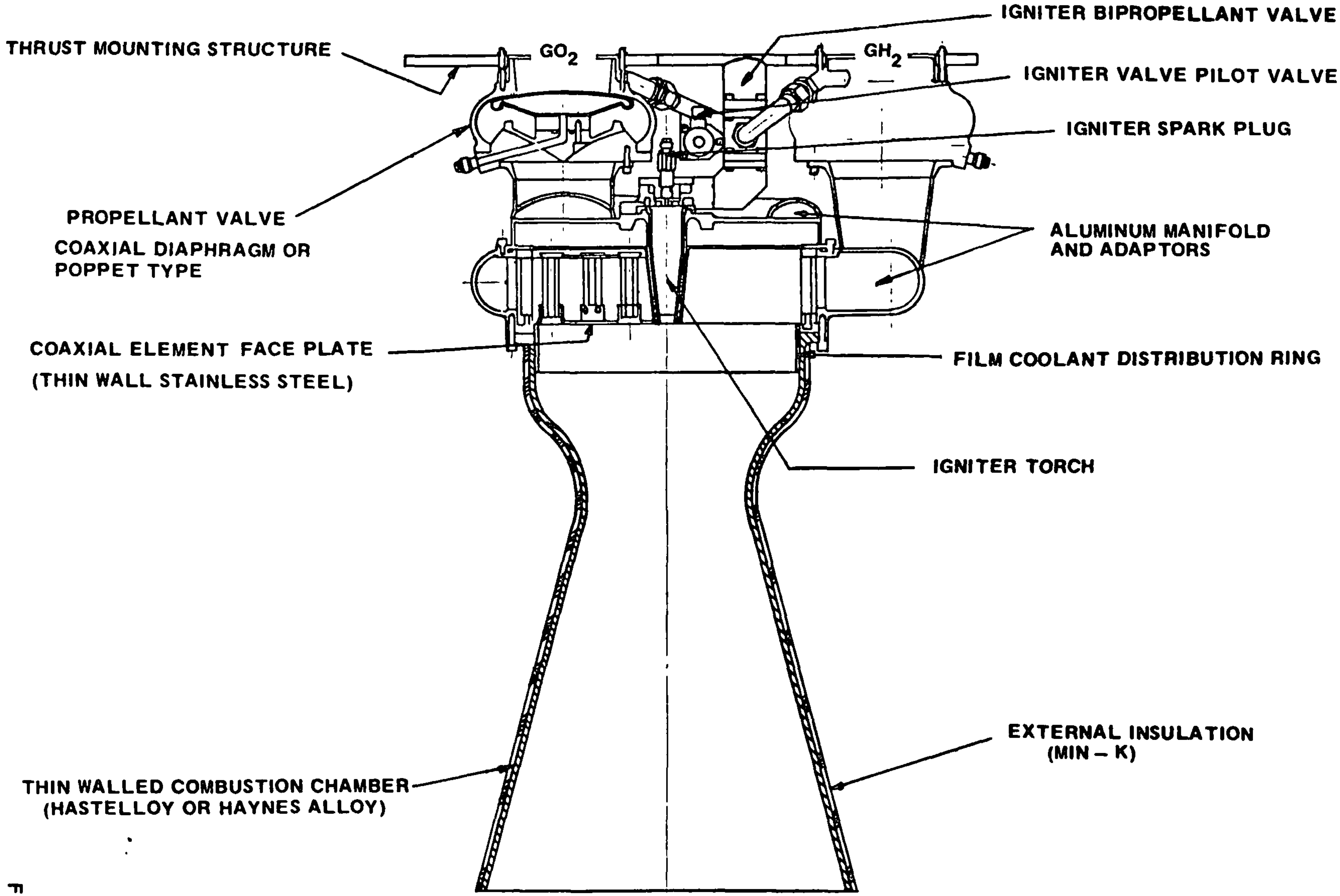
From tests of injector elements the minimum pressure drop between injector inlet and chamber for stable combustion was approximately 15 percent (2 lb/in^2).

Delivered specific impulse as a function of changes in design point variables was identified using the JANNAF standardized performance evaluation program. This technique identified the changes in specific impulse losses as a function of all combinations of operating variables. The expected delivered performance was determined by computing theoretical performance, then subtracting the following calculated performance losses:

(1) reaction kinetics and energy release
(2) mixture ratio distribution
(3) boundary layer
(4) nozzle divergence, and
(5) film cooling

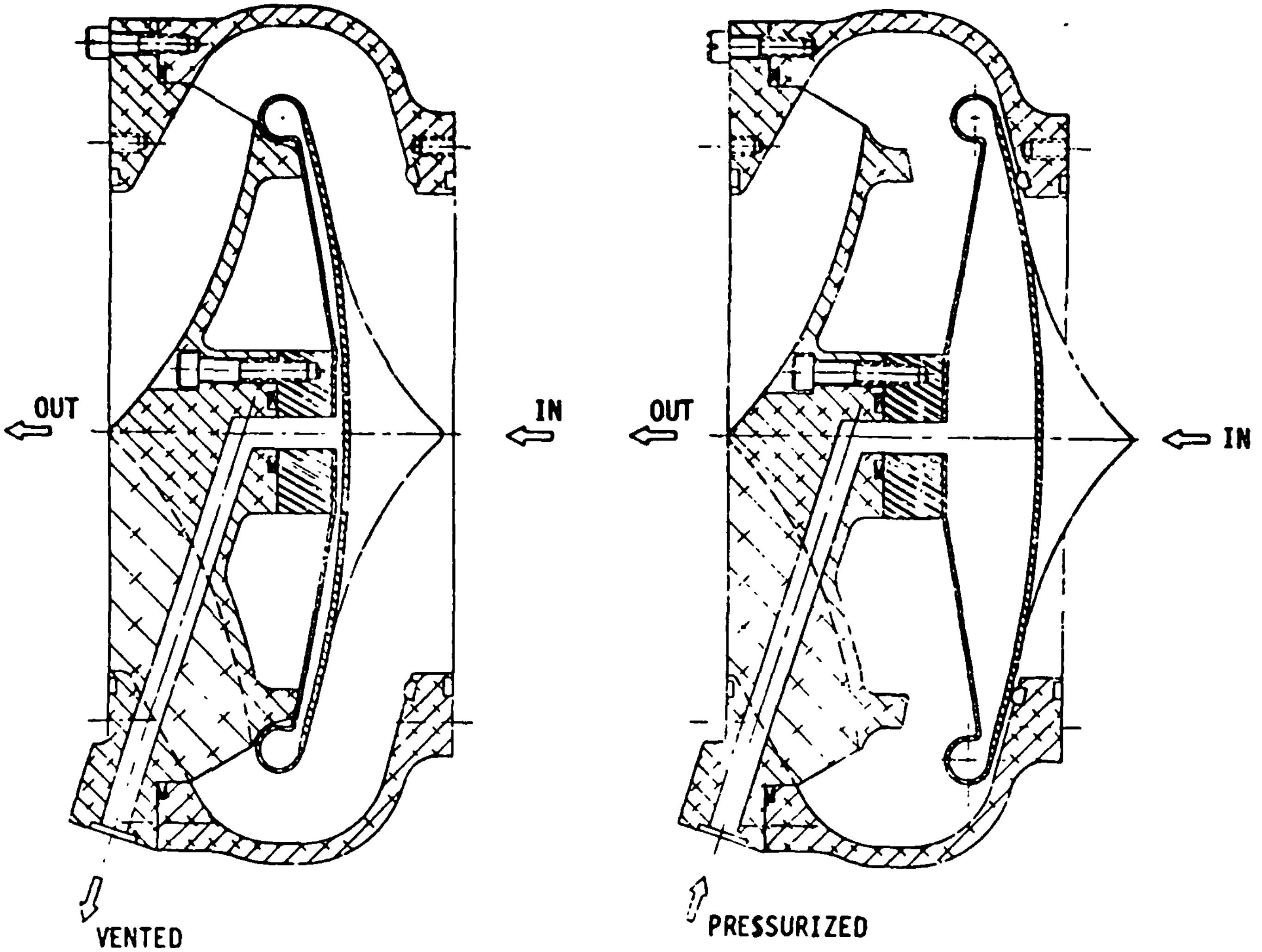
Supplemental film cooling is required, since all design operating points must meet APS cycle life requirements. These coolant requirements are also a function of the design point as shown in Figure D-4.

Coupling these cooling requirements and their corresponding performance losses with variations in the component losses, resulted in the performance trends shown in Figures D-5 to D-7. The most distinct variation results from changes



LOW PRESSURE APS ENGINE

FIGURE D–2



LOW PRESSURE ENGINE VALVE DESIGN

FIGURE D-3

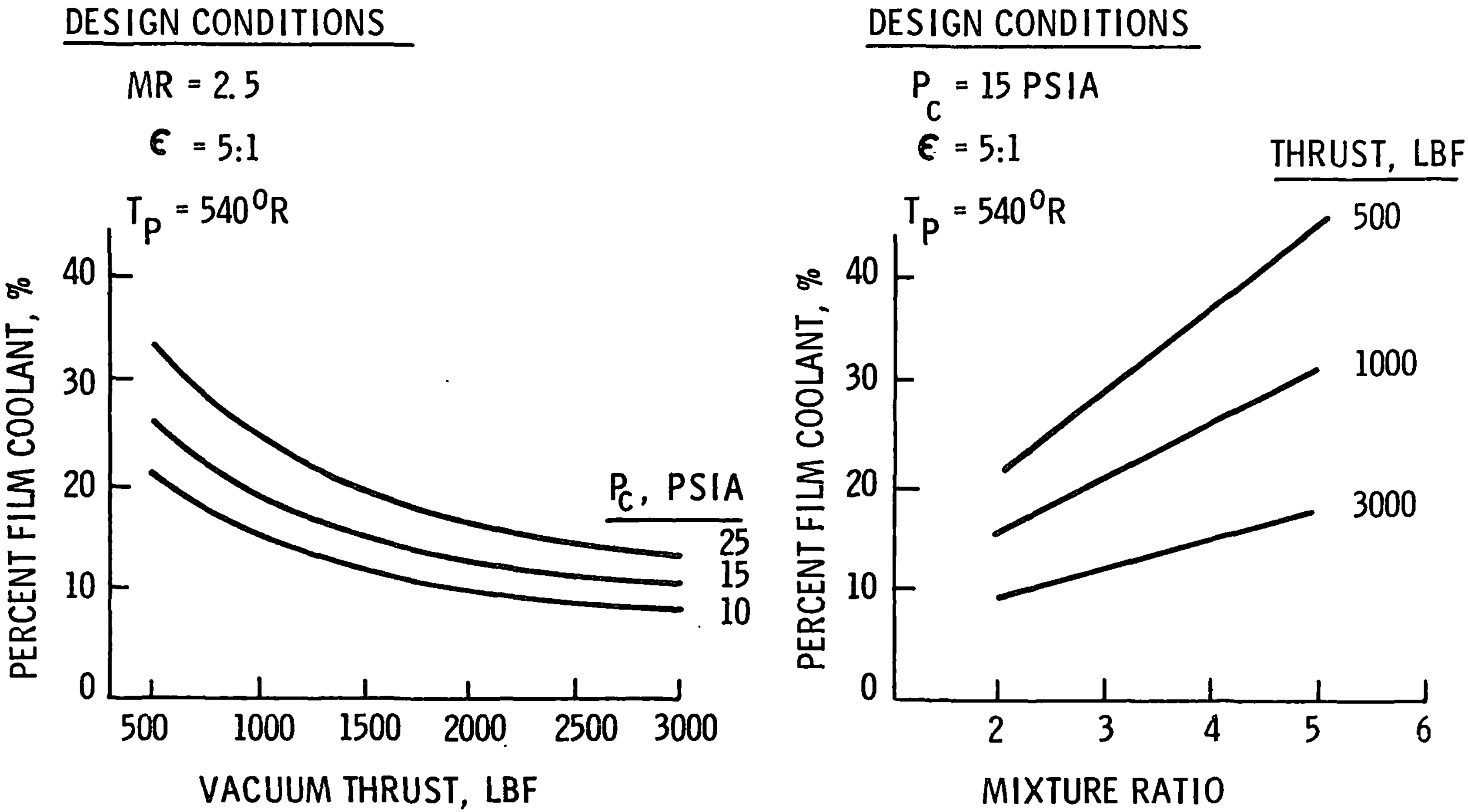


FIGURE D-4

TYPICAL DESIGN INFLUENCE ON FILM COOLANT REQUIRED

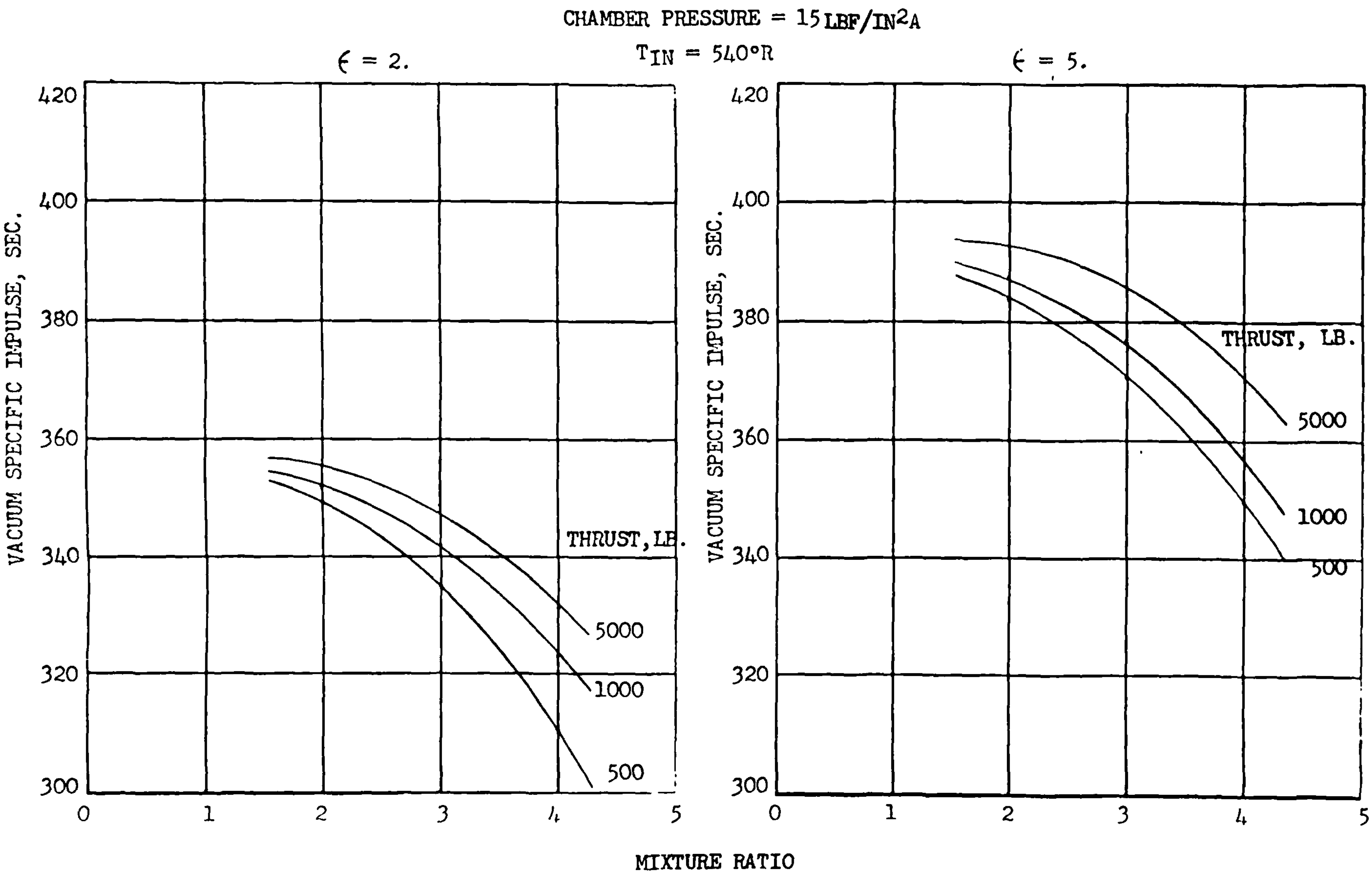


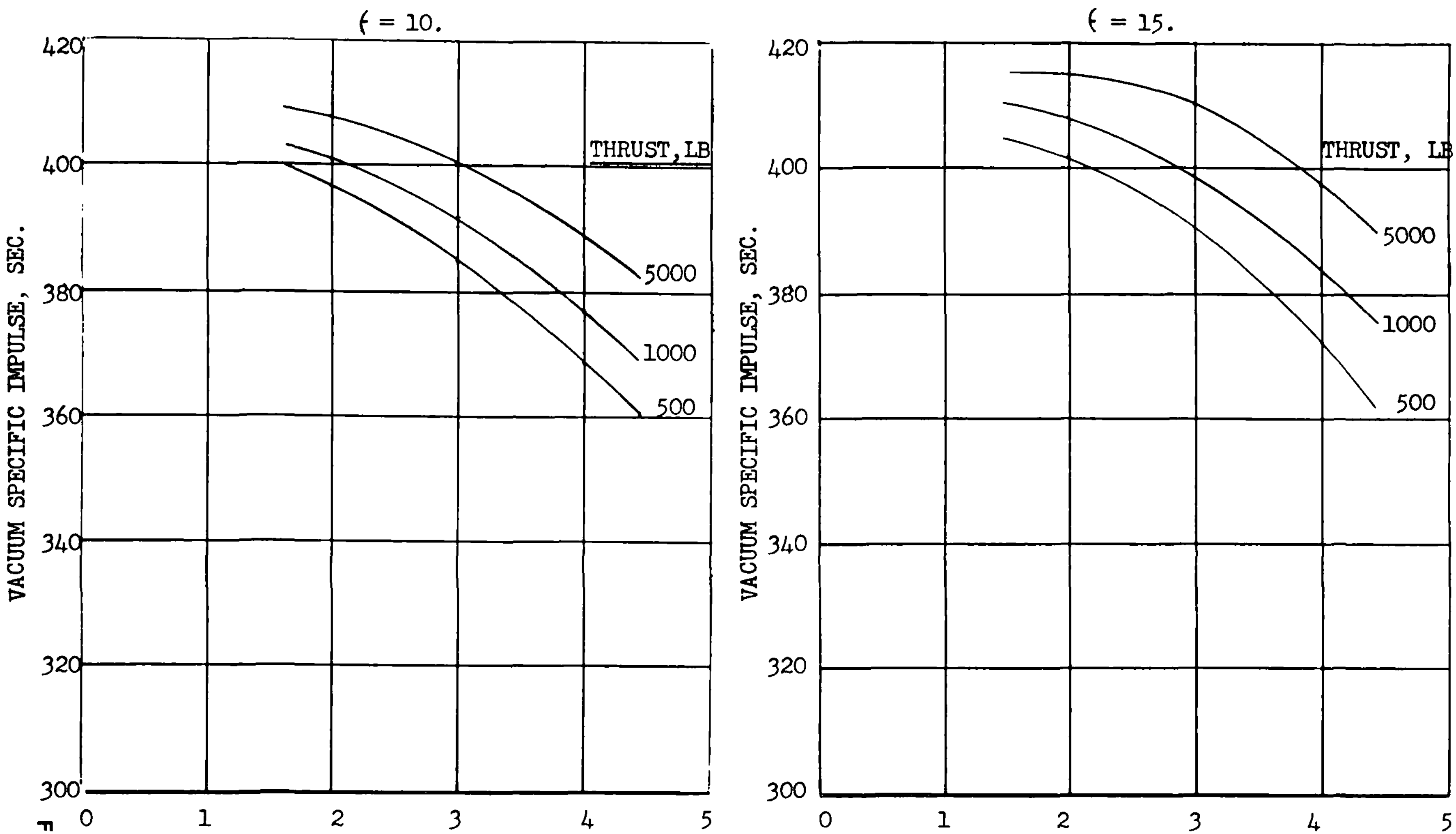
FIGURE D-5

LOW PRESSURE ENGINE PERFORMANCE

DESIGN MIXTURE RATIO INFLUENCE ON SPECIFIC IMPULSE

CHAMBER PRESSURE = 15 LBF/IN^2A

T_{IN} = 540°R



MIXTURE RATIO

FIGURE D-6

LOW PRESSURE ENGINE PERFORMANCE

DESIGN MIXTURE RATIO INFLUENCE ON SPECIFIC IMPULSE

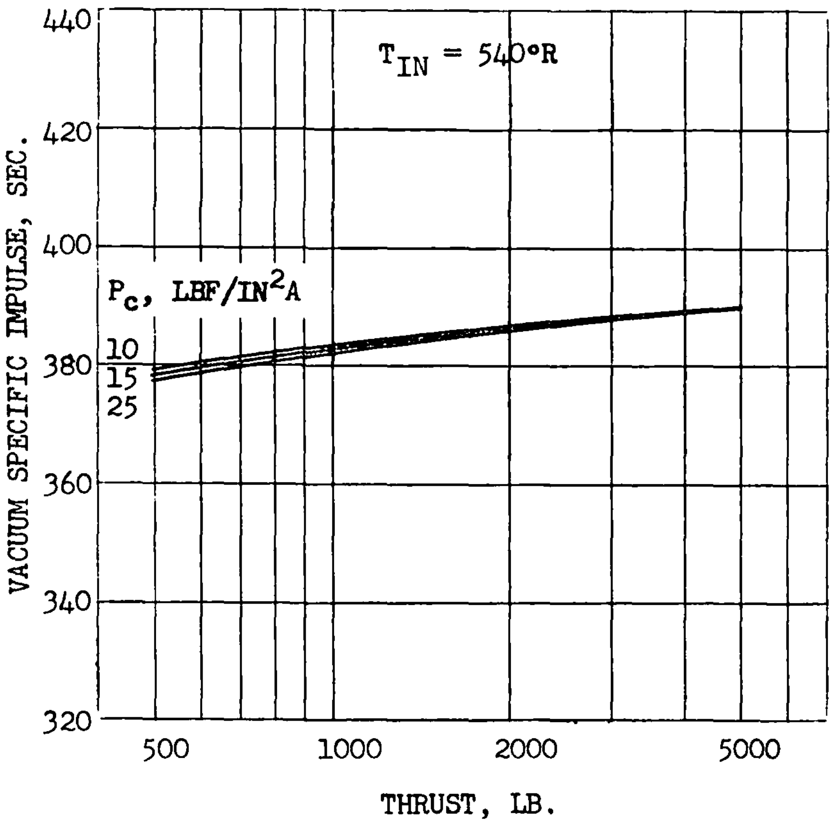
80% BELL NOZZLE

MIXTURE RATIO = 2.5

EXPANSION RATIO = 5.

CHAMBER PRESSURE = 15 LBF/IN^{2}A

DESIGN CHAMBER PRESSURE INFLUENCE
ON SPECIFIC IMPULSE



INLET TEMPERATURE INFLUENCE
ON SPECIFIC IMPULSE

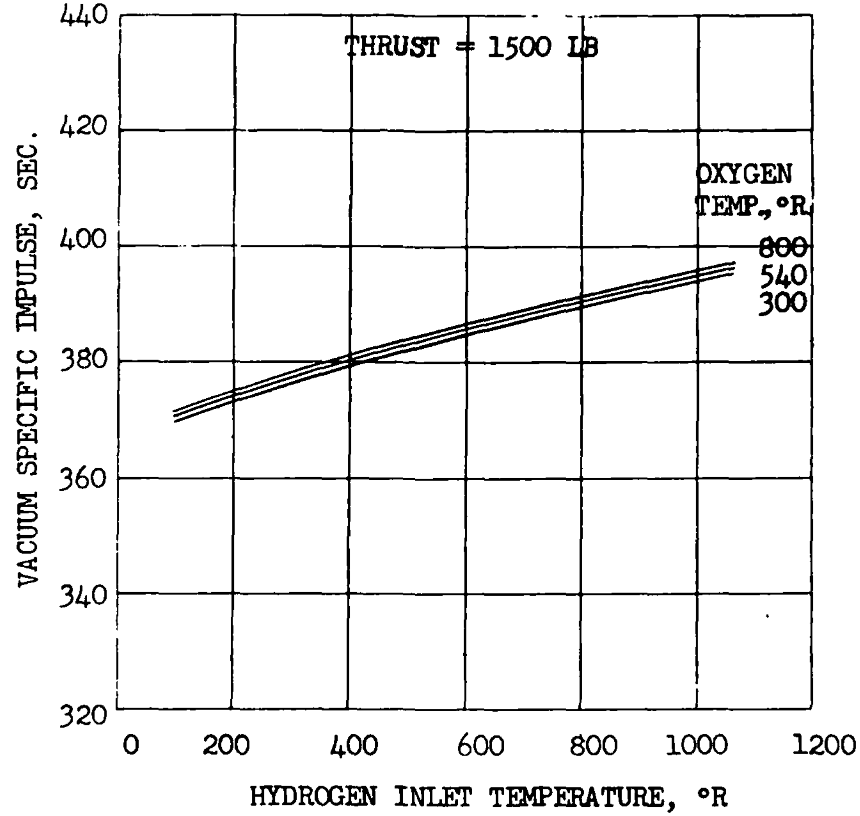


FIGURE D-7

LOW PRESSURE ENGINE PERFORMANCE

in design MR, Figures D-5 and D-6. As the engine operating point shifts from 2 to 4, a coolant flow increase is required to compensate for the higher combustor core gas recovery temperature. This performance loss increase, coupled with greater kinetics loss, reduced delivered performance at higher mixture ratios.

Performance gains with nozzle area ratio increase, Figures D-5 and D-6, resulted from changes in theoretical performance as the gas was further expanded.

Trends in performance with variations in chamber pressure (Figure D-7(a)) denote a compromise between cooling requirements and kinetics losses. Increasing chamber pressure usually increases performance by decreasing kinetics loss; however, this was offset by the influence of increased coolant flow requirement.

Thrust level changes (Figure D-7(a)) result in variations in propellant residence time (stay time) within the nozzle. Since nozzle length is directly proportional to chamber diameter at a given expansion ratio, stay time is increased. This increased length increases the time available for recombination and reduces the corresponding kinetics loss. In addition, coolant requirements are reduced, since chamber surface per unit mass flow is lower. These influences are combined to produce the overall change in specific impulse with chamber pressure.

Increasing propellant temperature (Figure D-7(b)) increased theoretical performance because of higher propellant enthalpy, though coolant flow changes were also required to provide an equivalent wall temperature.

The size and weight of the low pressure engines were very sensitive to design variables. Small variations in chamber pressure and expansion ratio represented a significant percentage change in design point and in corresponding component weight. Variation in physical properties as a function of thrust, chamber pressure, and area ratio are presented in Figures D-8 through D-10.

The heaviest engine component was the injector assembly. Realistic projections of injector weight have been made in the weight analysis assuming use of aluminum. Aluminum was also used in the weight analysis for the shutoff valves and injector assembly.

- EXPANSION RATIO = 2.0
- 80% BELL NOZZLE
- O/F = 3

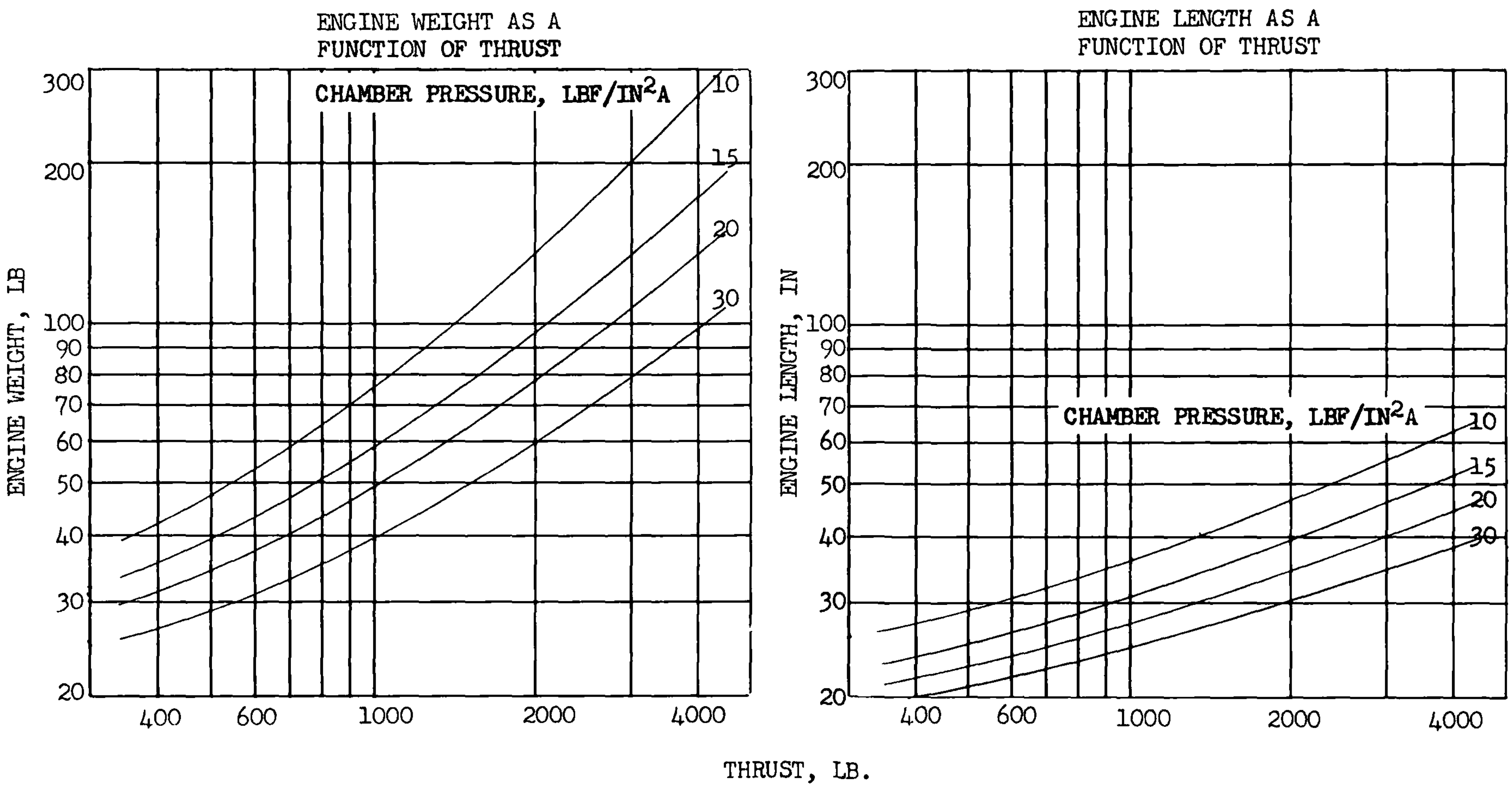


FIGURE D-8

LOW PRESSURE ENGINE PHYSICAL CHARACTERISTICS

- EXPANSION RATIO = 5.0
- 80% BELL NOZZLE
- O/F = 3

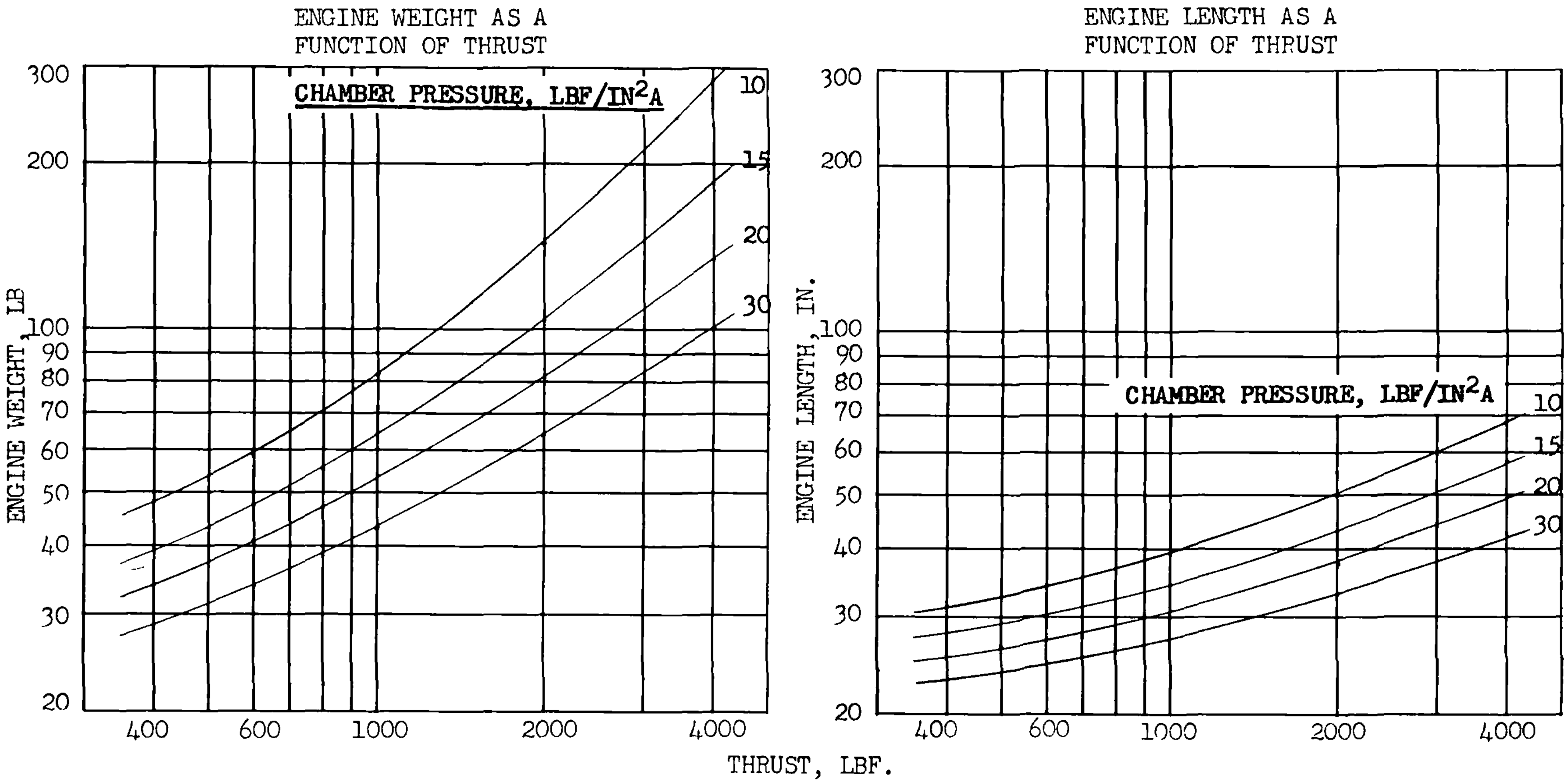


FIGURE D-9

LOW PRESSURE ENGINE PHYSICAL CHARACTERISTICS

- EXPANSION RATIO = 10.0
- 80% BELL NOZZLE
- O/F = 3

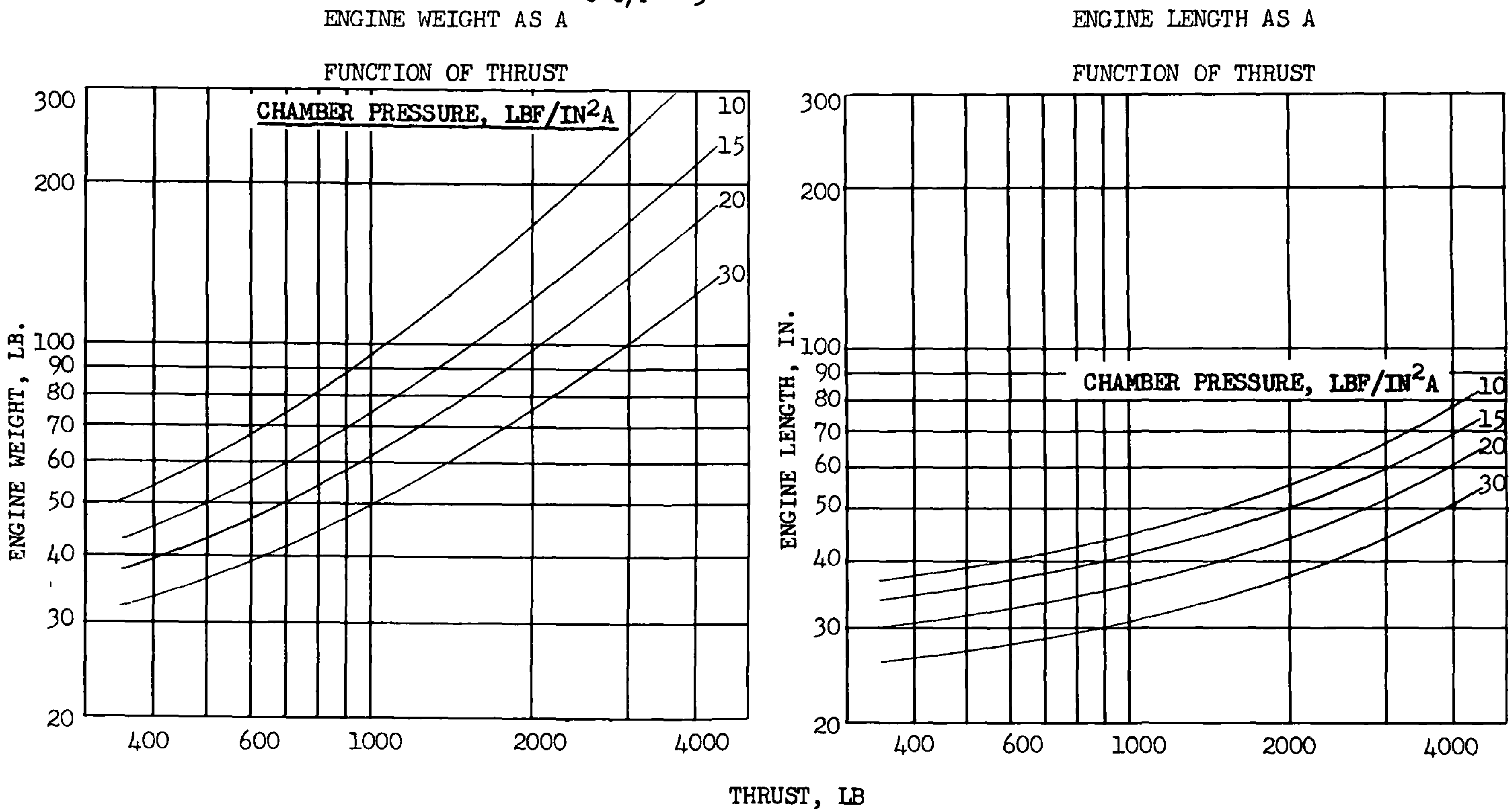


FIGURE D-10

LOW PRESSURE ENGINE PHYSICAL CHARACTERISTICS

D-3. GAS GENERATOR

The low pressure gas generator consists of cylindrical combustion chamber, injector assembly, electrical spark igniter, and associated propellant control valves, similar to those used in the low pressure engine. The gas generator is shown schematically in Figure D-11. The cylindrical chamber has an L/D ratio of 1.0 to 2.5; 1 for large diameters, and up to 2.5 for the smaller diameters. Designs utilizing an L/D ratio less than 1.0 were too short to allow homogenous combustion (uniform flame temperature). The chamber diameter was sized to obtain a combustion gas Mach Number of 0.2. Chambers contain a turbulence ring approximately halfway down from the injector face, to promote uniform mixing and combustion. The chamber body is constructed of A286 steel at a thickness of 0.050 in. The injector design is similar in materials and configuration to that used on the APS engines except for fewer elements and the allowance for the lower mixture ratio. The pressure drop across injector is approximately 20 percent of chamber pressure. Gas generator igniton response occurs in 30 to 40 ms. Allowing valve response time of 15 to 25 ms yields an overall gas generator response of less than 60 ms.

Gas generator performance, weight and size were calculated in a manner similar to that used for the low pressure engine. Performance and weight realtionships are shown in Figure D-12.

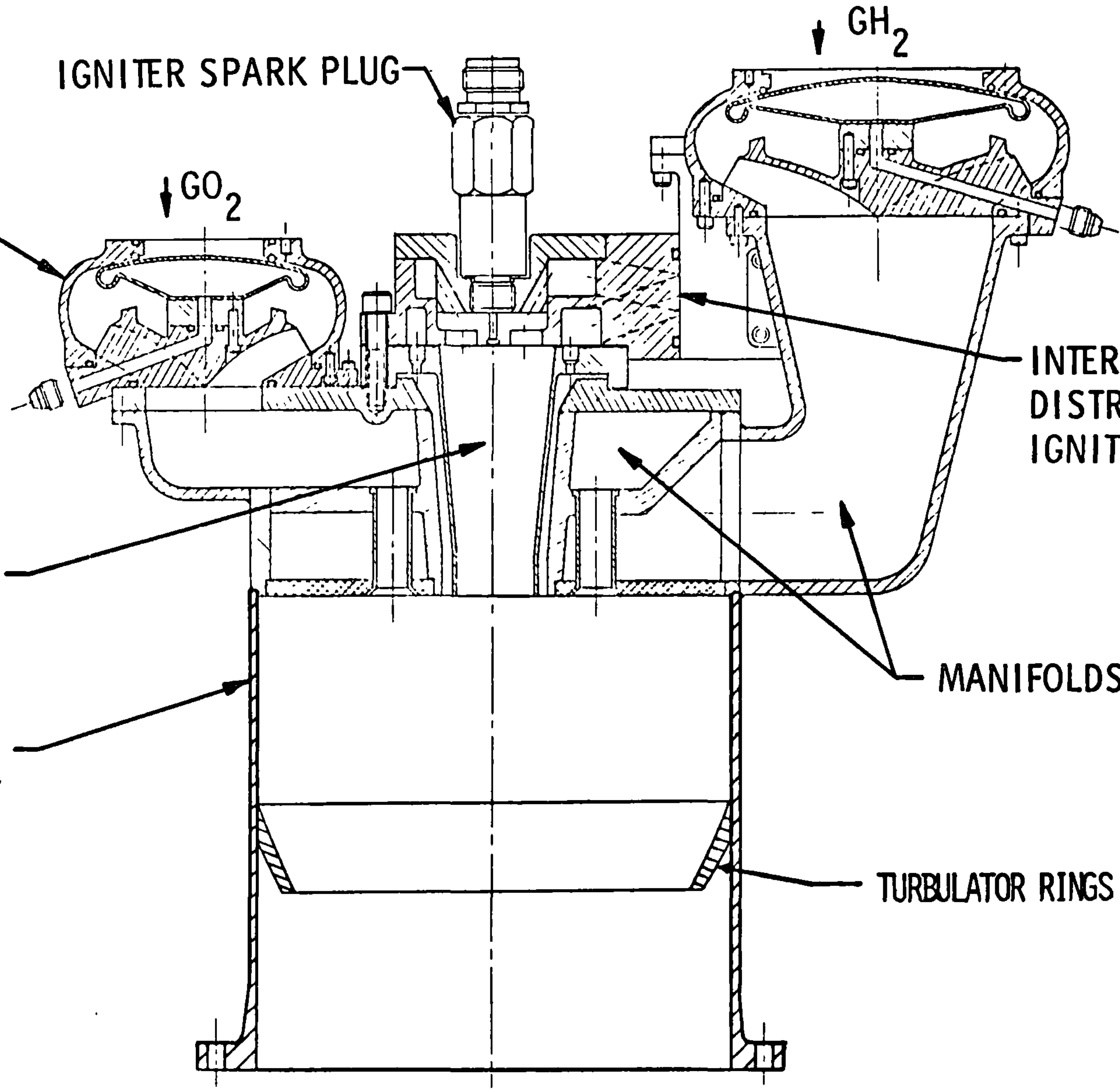
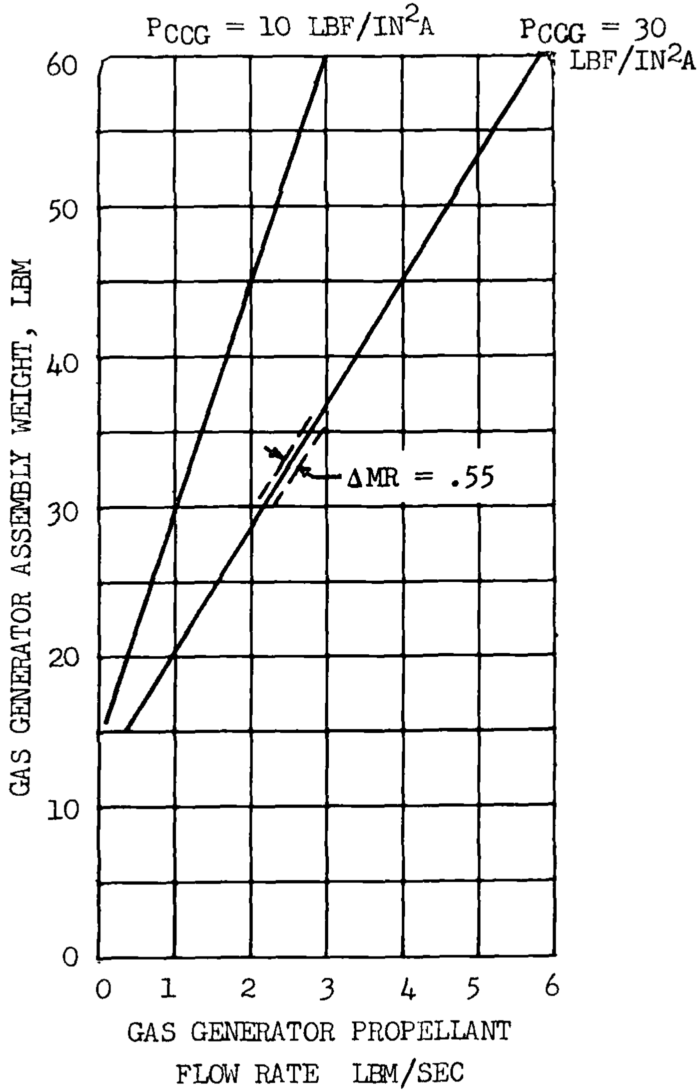
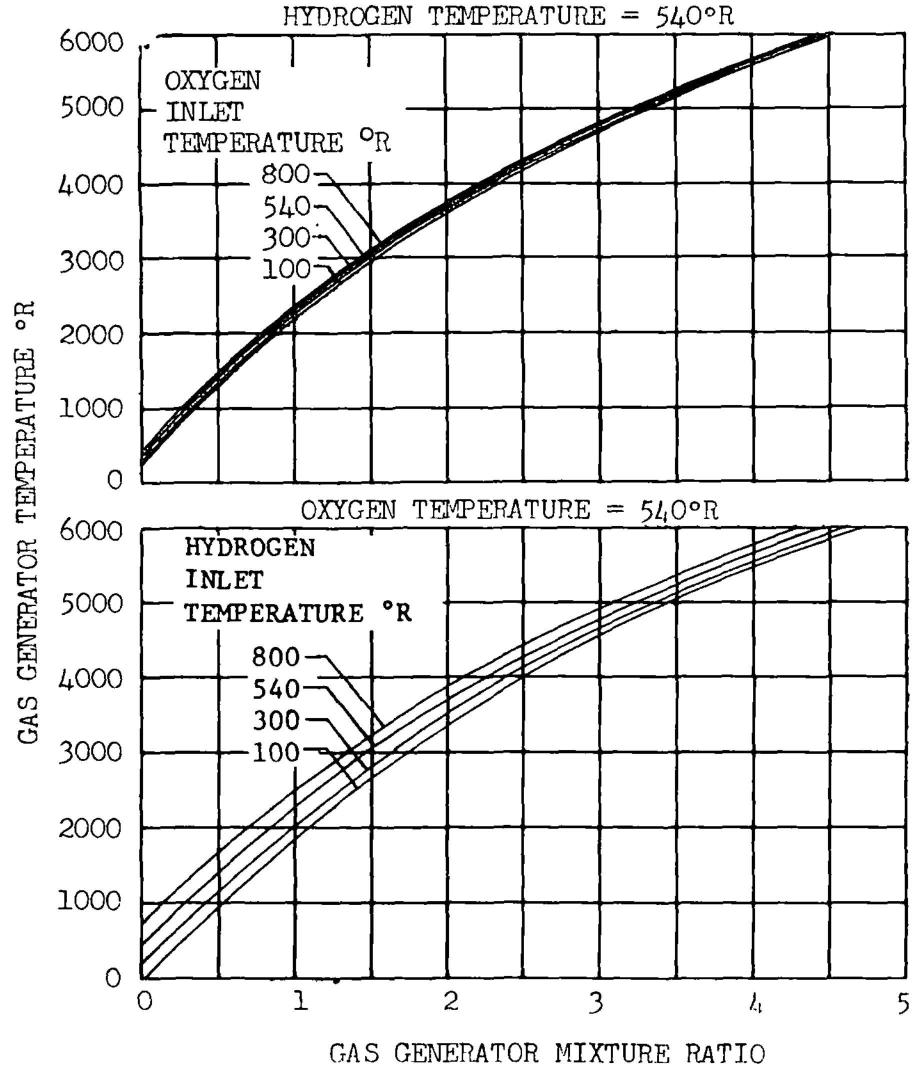


FIGURE D-11
LOW PRESSURE APS GAS GENERATOR



GAS GENERATOR ASSEMBLY WEIGHT
VERSUS
GAS GENERATOR PROPELLANT FLOW RATE



GAS GENERATOR PERFORMANCE

T_{GG} AS A FUNCTION OF MR_{GG}

FIGURE D-12

D-4. ACTIVE HEAT EXCHANGER

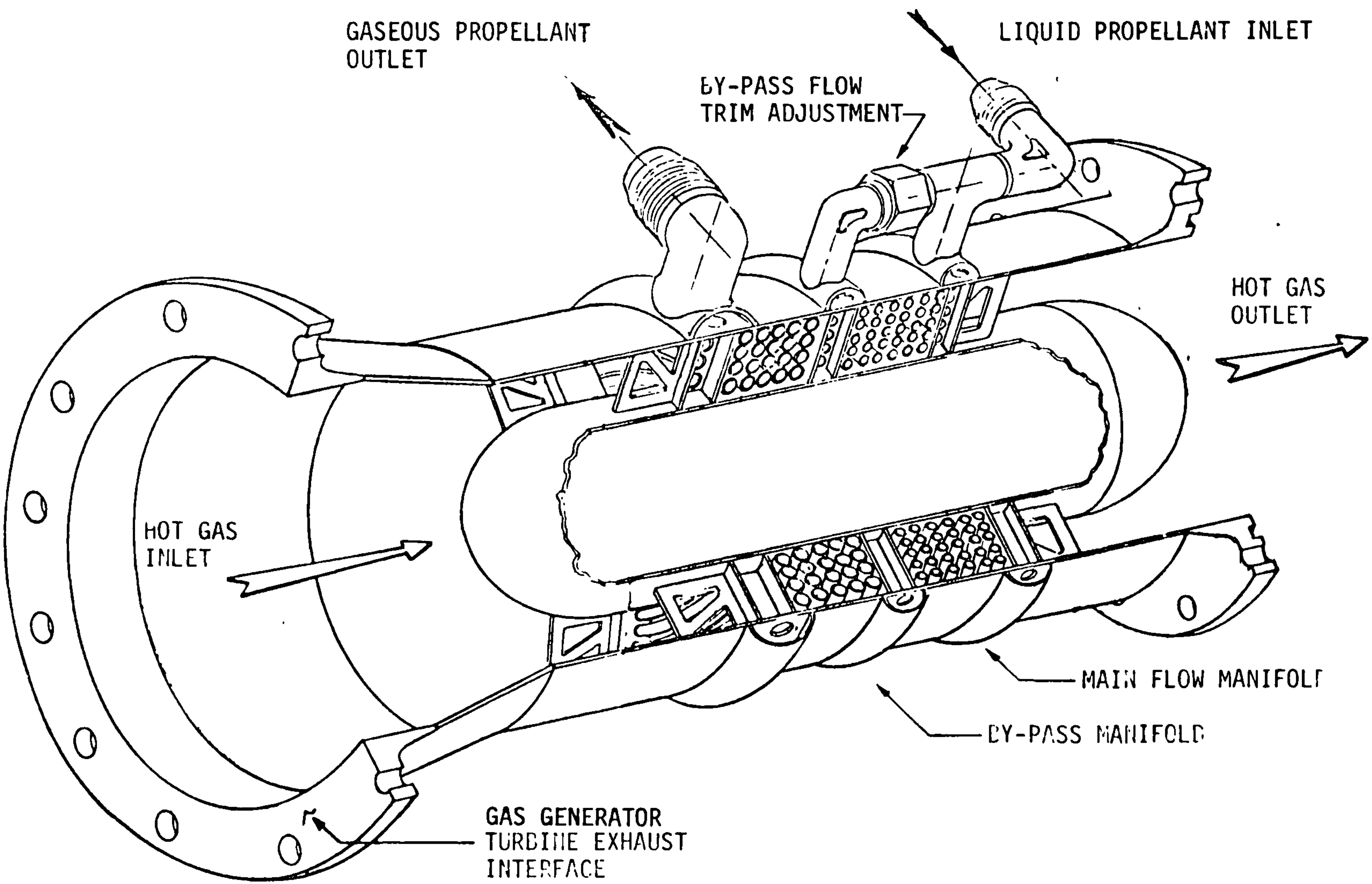
The basic design of the heat exchanger is a concentric helical tube bundle contained within a cylindrical shell as shown in Figure D-13. Hot combustion products from the gas generator are passed through the shell, heating the tube bundle through which liquid propellants flow. As propellant vaporizes and changes density, it passes through a center manifold and into another bundle of larger diameter tubes, allowing the fluid density to change without producing a large pressure drop across the tube length. A representation of the flow through the heat exchanger is given in Figure D-14.

The helical tube concept required that some void space exist in the center of the tube bundle because of practical minimum tube bending radius considerations. The helical coils are supported between manifolds by longitudinal plate baffles, which serve to position the center diffuser and strengthen the assembly. Propellant is distributed to, and collected from, the tubes by means of multiple radial manifolds symmetrically spoke-mounted in a plane normal to the hot gas flow. This multimanifold arrangement possesses many advantages; it allows the thermal expansion of the tubes during operation, exposes each row of tubes to an identical hot gas environment, and improves accessibility to the individual tubes for ease of fabrication. AISI 347 stainless steel was selected for the heat exchanger tubing because of its strength, fabrication characteristics, and heat transfer capability. Propellant inlet flow control devices were used at the inlet to ensure flow and pressure stability.

Operating limits for the heat exchanger are tabulated in Figure D-15. The indicated nominal design point conditions were used as baseline for the heat exchanger. In designing the heat exchanger for nominal and off-nominal operation, the limiting criteria (maximum exit flow Mach Number of 0.2 and minimum number of tubes for acceptable pressure variation within the length of a tube) determined heat exchanger tube sizing. The number of manifolds used was based on engineering judgement for reasonable packaging. Space between tubes was held at 0.15 in. which, in turn, determined coil pitch, number of tube rows, and all external and internal diameters of the heat exchanger.

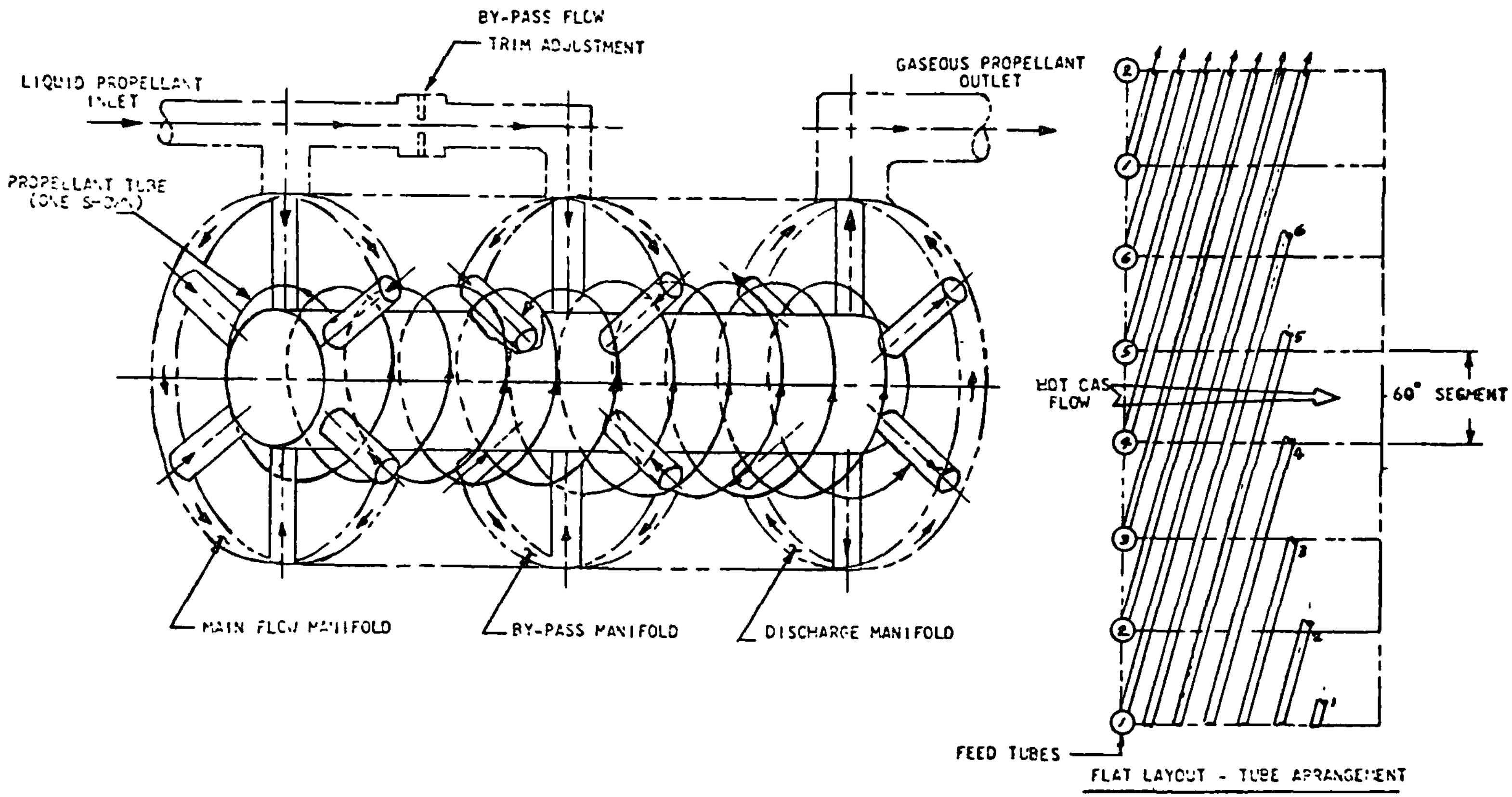
With the aforementioned data (number of tubes, sizes, manifolds, spacing, hot gas flow rate, core diameter, thicknesses, and design criteria selected for each case) a thermal analysis was conducted to determine tube bundle length.

AUXILIARY PROPULSION PROPELLANT CONDITIONING SUBSYSTEM



HEAT EXCHANGER

FIGURE D-13



FLOW SCHEMATIC-HEAT EXCHANGER

FIGURE D-14

	HYDROGEN			OXYGEN		
	Min.	Max.	Nominal Design Point	Min.	Max.	Nominal Design Point
P_{in_c} (LBF/IN^2A)	60	60	60	60	60	60
T_{in_c} (°R)	40	40	40	160	160	160
T_{out_c} (°R)	200	500	200	300	600	300
W_c (lb/sec)	.5	12	5	2	30	10
$P_{in_{hot}}$ (LBF/IN^2A)	10	30	30	10	30	30
$T_{in_{hot}}$ (°R)	1600	2500	2000	1600	2500	2000
$T_{out_{hot}}$ (°R)	700	1000	900	700	1000	900

OPERATING LIMITS FOR LOW PRESSURE HEAT EXCHANGERS

FIGURE D-15

Heat balances were made for each design point to determine the amount of hot gas required to achieve desired cold and hot side operating conditions. Equations for film coefficients, two phase flow conditions, etc. were selected on the basis of engineering judgement for applicability and validity of correlations within specific operating regimes. In developing the mathematical model, the following assumptions were made:

(1) angular variation of hot gas properties was negligible, especially in the multimanifold design

(2) radial variation in cold flow tube diameter was neglected, and an average tube located at an average diameter replaced the tube bank

(3) due to the high temperature differences, a parallel flow analysis was used, but would not differ significantly from a counter flow analysis

(4) in performing a step by step heat, mass, and momentum balance down the tube, film coefficient and friction factor were evaluated using the properties at the section inlet

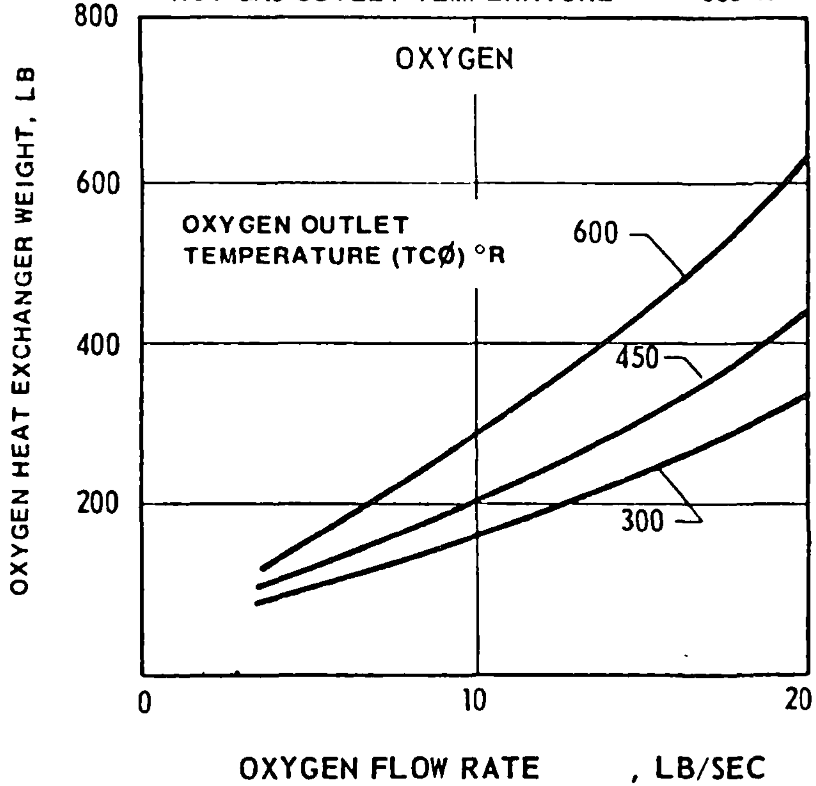
(5) wall resistance caused negligible wall temperature drop

(6) no ice or liquid water formed on the hot side tube walls

With the overall tube length to achieve desired outflow conditions established, overall heat exchanger weight and size were defined. The assembly weight was arrived at by detailed calculations of tube bundle, manifolds, baffles, core, and shell weights. Heat exchanger weights, lengths, and diameters are plotted as functions of specified independent variables in Figures D-16 and D-17.

DESIGN CONDITIONS

OXYGEN INLET TEMPERATURE	=	160°R
OXYGEN INLET PRESSURE	=	60 LBF/IN^2 A
OXYGEN OUTLET PRESSURE	=	40 LBF/IN^2 A
HOT GAS INLET TEMPERATURE	=	2000°R
HOT GAS INLET PRESSURE	=	30 LBF/IN^2 A
HOT GAS OUTLET TEMPERATURE	=	900°R



DESIGN CONDITIONS

HYDROGEN INLET TEMPERATURE	=	40°R
HYDROGEN INLET PRESSURE	=	60 LBF/IN^{2}A
HYDROGEN OUTLET PRESSURE	=	40 LBF/IN^{2}A
HOT GAS INLET TEMPERATURE	=	2000°R
HOT GAS INLET PRESSURE	=	30 LBF/IN^{2}A
HOT GAS OUTLET TEMPERATURE	=	900°R

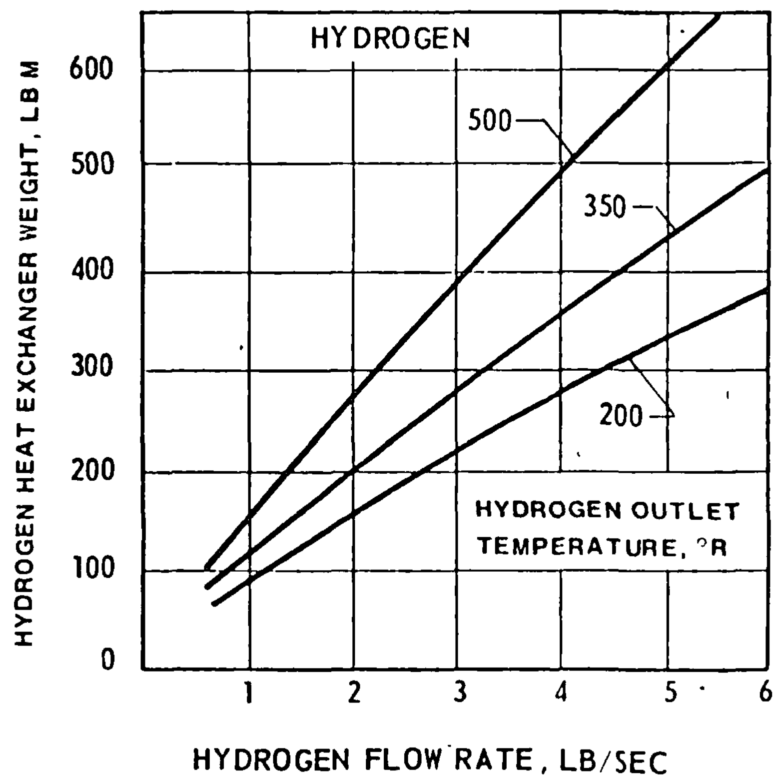


FIGURE D-16

LOW PRESSURE
HEAT EXCHANGER WEIGHT

DESIGN CONDITIONS
HYDROGEN INLET TEMPERATURE = 40°R
HYDROGEN INLET PRESSURE = 60 LBF/IN^2A
HYDROGEN OUTLET PRESSURE = 40 LBF/IN^2A
HOT GAS INLET TEMPERATURE = 2000°R
HOT GAS INLET PRESSURE = 30 LBF/IN^2A
HOT GAS OUTLET TEMPERATURE = 900°R

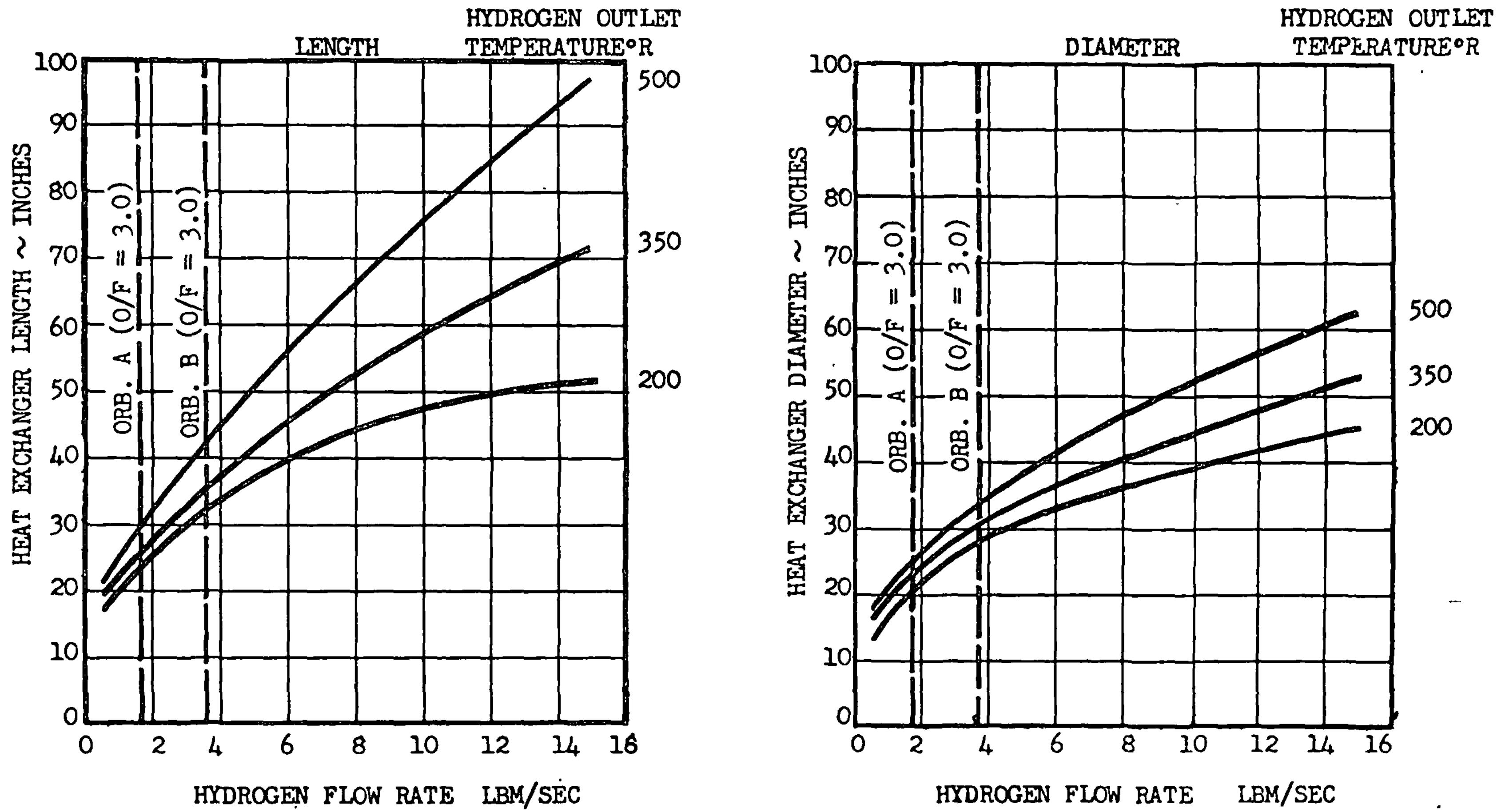


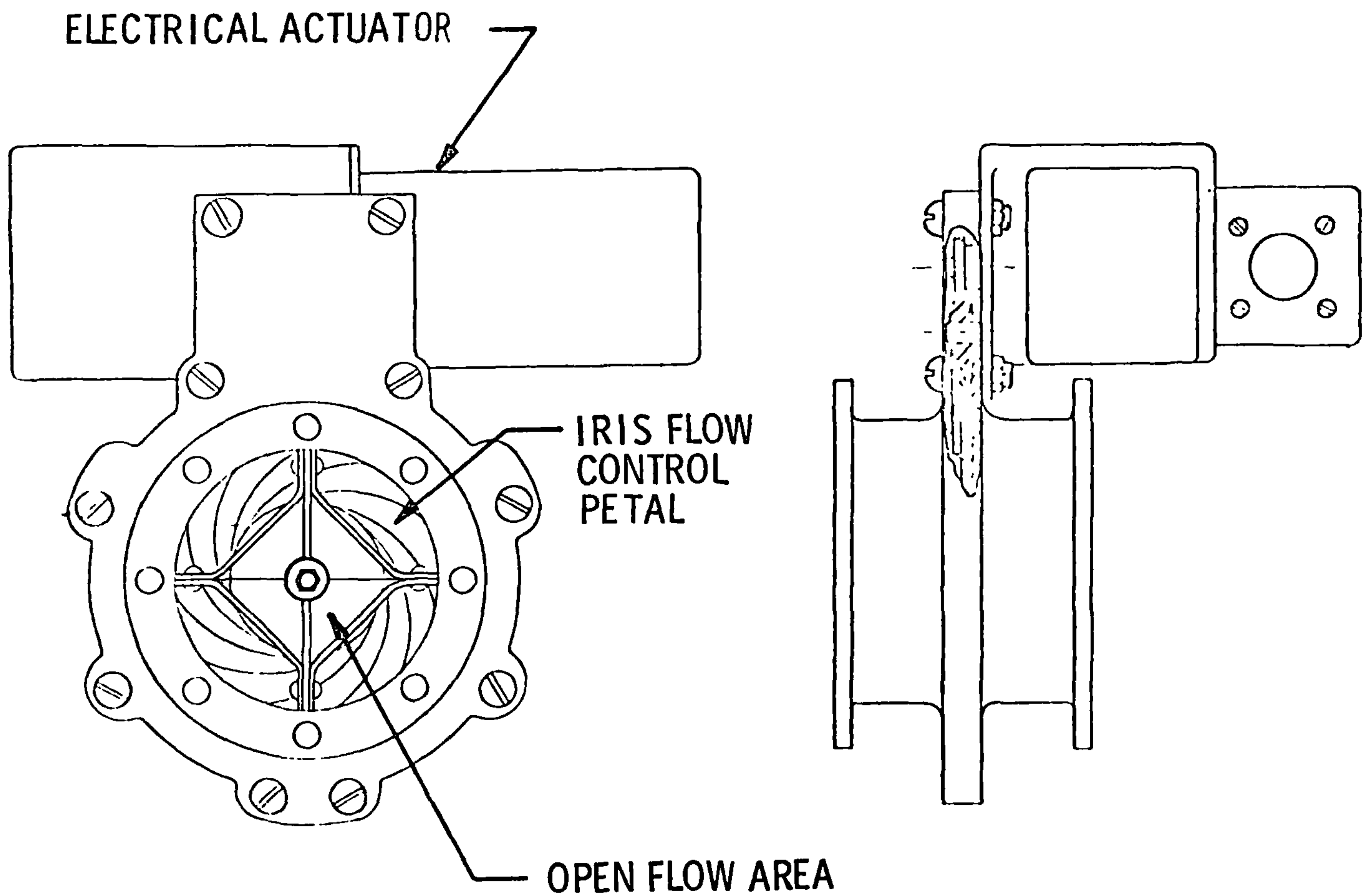
FIGURE D-17

HYDROGEN HEAT EXCHANGER LENGTH AND DIAMETER

D-5. PRESSURE REGULATORS

The regulator design, illustrated in Figure D-18 possesses a series of overlapping segments that are moved to control propellant flow area. The general configuration is similar to the aperture control device in a camera. Since this type of unit does not have a positive shutoff ability, it would be placed downstream of a shutoff valve. The iris-type regulator would be driven by an actuator in response to an electrical signal from a downstream sensor. The iris would not fully close; rather, it would maintain a given setting when the engine valves were signaled to close. When valves were signaled to open, the regulator feedback loop would again function, establishing regulator flow area in response to electrical signals.

The iris regulator orifice size was calculated for flow conditions of 25 lbf/in^2a inlet pressure, 4 lbf/in^2 pressure drop across the regulator, and gas temperature of 500°R. Weights of regulators with different flow ratings were obtained from Lundy Electronics and Systems Inc., and are plotted in Figure D-19. These weights, and the equations, were adjusted to include the accompanying sensor and actuator weights.



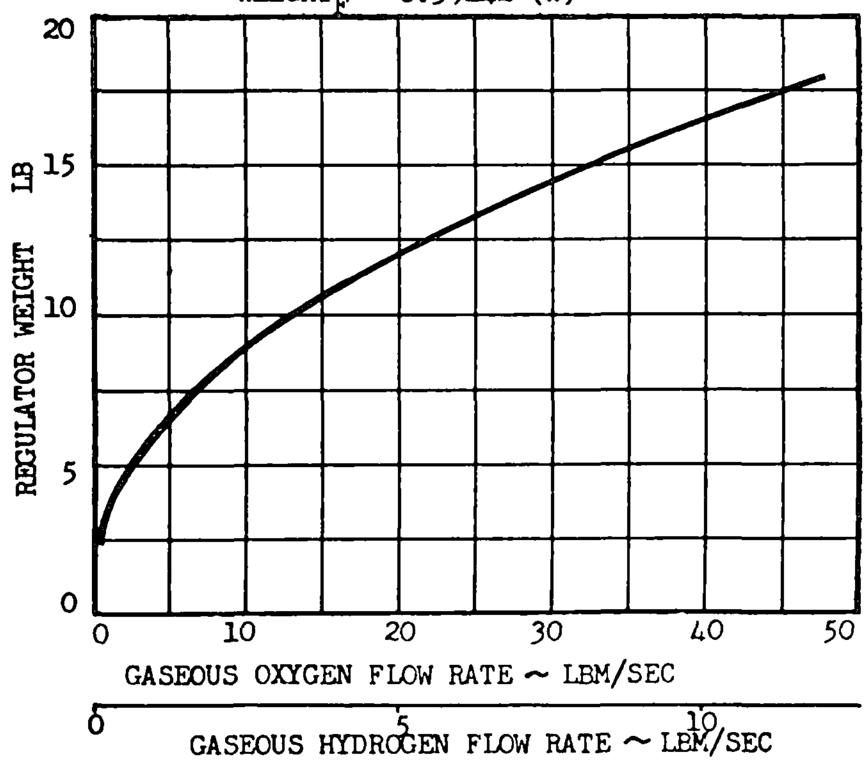
IRIS REGULATOR CONCEPT

FIGURE D-18

WEIGHT EQUATION

$WEIGHT_{OX} = 3.66081\,(W)^{.401987}$

$WEIGHT_{F} = 6.39142\,(W)^{.401987}$



IRIS TYPE
$P_{INLET} = 25$ LBF/IN^{2}A
$\Delta P = 4$ LBF/IN^2
$T = 500°R$

FIGURE D-19

PRESSURE REGULATOR WEIGHT VS FLOW RATE

D-6. PROPELLANT TRANSFER PUMP

It is possible to transfer the cryogenic propellants (LH_2 or LO_2) from the liquid storage tank, through the heat exchanger to the boost tank by means of electrically driven pumps. These pumps can be tank mounted, and operate with as little as 0.5 ft NPSH. Comparative performance data are given in this section for systems that deliver a 20 lbf/in^2 pressure rise at flow rates of 2 lb/sec liquid hydrogen and 8 lb/sec liquid oxygen.

Electrical driven pumps can be operated by direct coupling to a high speed ac or dc electric motor. The hydrogen pump motor would operate with motor and bearings submerged in liquid hydrogen for cooling. Similarly, for oxygen pump cooling, the rotor would be cooled by gaseous helium, while liquid oxygen would cool the stator and the gaseous helium. Data for these pumps are tabulated in Figure D-20.

	LO_2		LH_2	
Pump Flow	8 lb/sec		2 lb/sec	
Pump ΔP (LBF/IN^{2}A)	20		20	
Electric Power	AC	DC	AC	DC
Frequency (cps)	400	--	400	--
Voltage	56 min.	28	56 min	28
Wattage	250	250	2800	2800
Shaft Speed (rpm)	7300	7300	22000	22000
Start Time (sec)	--	--	0.15	0.25
Envelope Dia. (In)	8	8	8	8
Envelope Length (in)	12.6	12.6	14	16
Weight (lb)	13	14.3	16	18
Model*	144-666M	144-666M	144-668M	144-668M

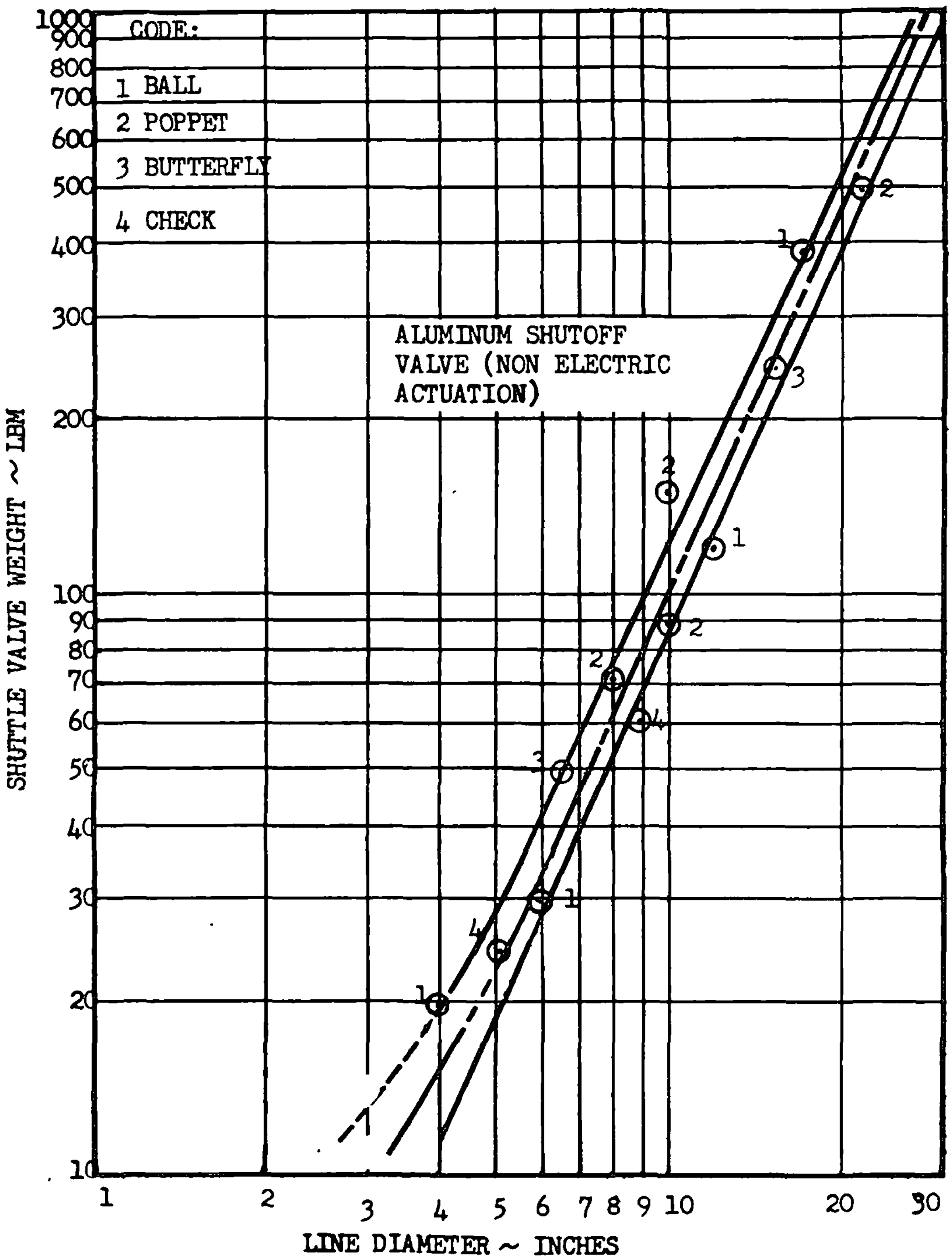
*PESCO Products Model Designation

FIGURE D-20

MOTOR DRIVEN TRANSFER PUMPS

D-7. PROPELLANT SHUTOFF VALVE

Shut-off valves considered for the low pressure APS included ball, butterfly, poppet, blade, and diaphragm varieties. Valve weight data (presented in Figure D-21, demonstrate that, from a weight standpoint, the type of shutoff element is not an overriding consideration, since valve weight is primarily a function of flow area or line size. Therefore, for APS Subtask A studies, selection of shut-off valve configuration was not made; instead, the data of Figure D-21 were employed in subsystem weight evaluations.



SHUTOFF VALVE WEIGHT

FIGURE D-21

APPENDIX E: RESIDUAL PROPELLANT AVAILABILITY

E-1. INTRODUCTION

At the times of booster and orbiter main engine shutdown, there are significant quantities of oxygen and hydrogen residual liquids and gases remaining in the main engine tanks. A summary of propellant residual quantities remaining in the Vehicle A booster and orbiter at engine shutdown is presented in Figure E-1. These residual propellants are assessed entirely against the main engine system, and if a portion of the propellants could be utilized by the APS to satisfy impulse requirements, APS weight might be reduced substantially.

Not all residual propellants are available for APS usage. After main engine shutdown, heat transfer from the surrounding vehicle structure and vaporization of residual liquids will increase main engine tank pressures to vent levels, and residual propellants will be lost through venting unless used by the APS. Thus, determination of the amount of residuals that can be credited to the APS, requires an analysis coupling combined effects of heat transfer into the main engine tanks, residual liquid vaporization, and APS propellant usage. The greatest uncertainty in this analysis is the heat transfer and vaporization rates associated with residual tank liquids. Since vaporization occurs in a very low gravity environment, evaluation of vaporization rates requires modeling of fluid motion and heat transfer characteristics in a low gravity environment. Such analytical techniques were not available. Instead, the approach selected for this study was to utilize data correlations for existing launch vehicles. Liquid vaporization rates experienced on previous launch vehicles were correlated to develop a model to predict equivalent propellant motion and heat transfer effects for known mission and vehicle characteristics. This model was then applied to the shuttle vehicle design and mission in order to identify expected vaporization rates.

This appendix presents the correlation of booster vaporization rate data and its application to the low pressure APS. The following paragraphs provide a definition of the Saturn vehicles examined, and a discussion of data and results for both oxygen and hydrogen. A summary of equivalent models which best approximate fluid motion and heat transfer are provided. Finally, these results are applied to the space shuttle vehicle to predict the availability of liquid residuals for the low pressure APS.

	BOOSTER A		ORBITER A	
Liquid	O_2	H_2	O_2	H_2
Tank	1502	0	276	0
Engines (Not Available)	2184	176	397	32
Lines	11,116	455	1154	116
Propellant Utilization Error	0	3191	0	362
$\sum$ Liquid	14,802	3822	1827	510
Gas	12,400	3440	2160	595
Total Residuals	27,202	7262	3987	1105
% Liquid	54.5	53.5	46	46.2
Temperature, Vapor	300	200	300	200
Pressure, Vapor	35	45	35	45

*Residuals specified in lbm

FIGURE E-1

RESIDUALS AT CUTOFF - VEHICLE A*

E-2. SATURN DATA EVALUATION

Low gravity, orbital data were available for three booster configurations. These were: (a) Saturn-IV, (b) Saturn-IVB/IB, and (c) Saturn-IVB/SV. The Saturn-IV vehicle was the upper stage of the Saturn I, two-stage launch vehicle. This vehicle performed a single burn to earth orbit. The Saturn-IVB/IB vehicle was the upper stage of the Saturn IB, two-stage launch vehicle. The Saturn-IVB flight numbers for this booster configuration were in the 200 series. The Saturn-IVB/SV vehicle was the third stage of the Saturn V, three-stage, launch vehicle. This vehicle performed a two-burn mission; the first burn was to earth orbit, followed by an earth orbital coast and a second burn for trans-lunar orbit insertion. Flight numbers for this booster configuration were in the 500 series. The S-IVB/IB and S-IVB/V stages were of basically similar configuration.

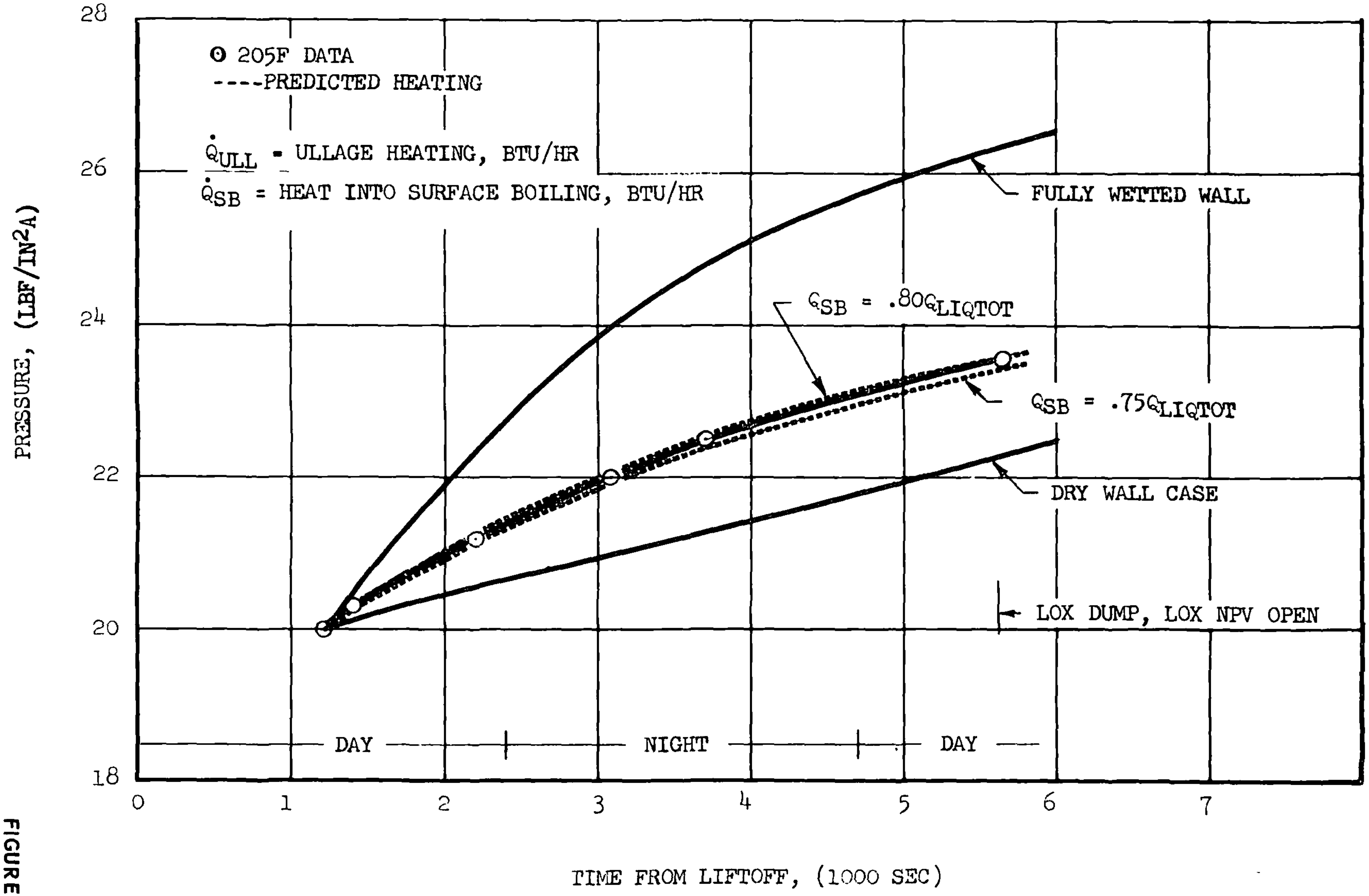
All S-IV and S-IVB flights were examined to obtain applicable data for the oxygen. Eight flights provided data suitable for study. Segments from 11 flights were examined to obtain data applicable to the hydrogen propellant. Six groups of these data were analyzed in depth, using one particularly well-instrumented group to correlate the other data. The following paragraphs provide a summary of the data obtained and a discussion of its interpretation.

E-2.1 Liquid Oxygen Evaluation - For the eight flights which provided LO_2 data suitable for study, heating was divided into two regimes. These were: (1) ullage gas heating or heating characterized by a vapor interface at the propellant tank wall and (2) liquid heating or heating characterized by a liquid interface at the propellant tank wall. This latter heating regime was further subdivided to identify the amount of liquid heating that should be allocated to propellant boil-off, and the amount associated with bulk liquid heating. Applicable data were evaluated to determine the amount of liquid/gas heating, wetted wall surface area, and percentage of liquid heat transfer associated with propellant boil-off.

To evaluate the liquid oxygen tank conditions during low gravity coast, the tank conditions at the beginning and end of the coast were established as closely as possible. For example, for one typical flight (Vehicle 204), the following conditions were established. After main engine cutoff, the stage was pitched for alignment with the local horizontal. Subsequently, the following tank conditions were achieved: LO_2 bulk temperature 161° - 162°R, ullage gas temperature 160°R and ullage pressure 19 lbf/in^2a. These were used as the initial conditions for evaluation of LO_2 characteristics. The conditions at the end of a two-hour coast were: LO_2 bulk temperature 164° - 165°R, ullage gas temperature 165° - 170°R and ullage pressure 26.3 lbf/in^2a. The observed pressure rise could not be due to ullage heating alone, as that would require an ullage temperature of 180°R. Bulk boiling of the liquid also could not have occurred, since bulk liquid temperatures never reached saturated conditions. Thus, a combination of liquid heating, liquid vaporization, and ullage gas heating was required to explain the measurements. To determine the appropriate balance between vaporization and ullage heating, a computer program was used to investigate the influence of various parameters affecting pressure and temperature within the vehicle. Using this approach, the balance between ullage heating and vaporization, which best satisfied end conditions, was deduced. For the data point in question, the average bulk heating rate for the residual liquid was approximately 1500 Btu/hr. Heat input rates required to vaporize sufficient liquid for the observed pressure rise rates were also determined. These varied between 5,000 and 15,000 BTU/hr. These results indicated that between 75 to 90 percent of the total heat transferred from the tank to liquid oxygen resulted in vaporization. The remainder was associated with bulk liquid heating.

As another example, Figure E-2 shows the pressure rise as a function of time for flight 205. Calculations of the pressure rise rate for two extreme cases are also shown. Because of incomplete ullage temperature instrumentation, it was necessary to estimate flight 205 heating rates using flight 204 data. The extreme cases considered were: (1) a fully wetted wall in which all of the heating was directly applied to the liquid; and (2) a dry wall situation in which all heat transfer was delivered to the ullage gas. These extremes are shown to bound the actual data. A curve which best satisfied observed data was deduced for this flight (in the same manner as described above) by assuming that 75 percent of total liquid heating was associated with surface boiling. As shown, this provides a very good model of the results.

Other flight data indicated different pressure profiles during coast. For example, the rate of pressure rise as a function of time from insertion was observed to vary from flight to flight. This was partially correlated with sun angle, which varied for different vehicles because of different launch times during the day. Other factors contributing to differences were the amount of liquid residuals, ullage gas composition (helium and oxygen), ullage temperature, and extent of orbital maneuvers.



LOX TANK ULLAGE PRESSURE (S-IVB-205, LOW-G COAST)

FIGURE E-2

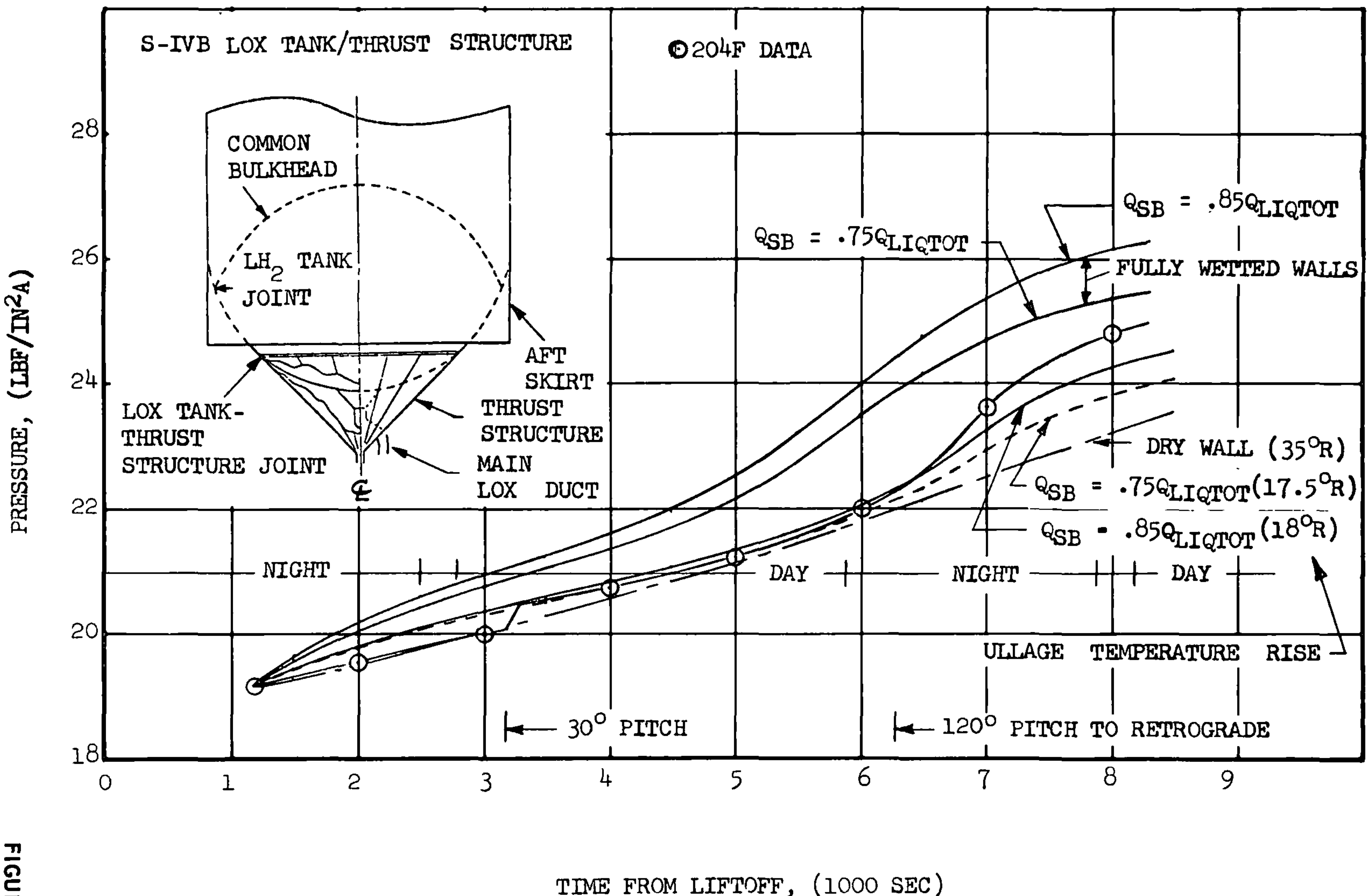
The pressure history of flight 204, shown in Figure E-3, clearly shows the singular effect of orbital maneuvers. Pitch maneuvers were executed at 3200 and at 6300 seconds. These are believed to have positioned liquid over tank support structure which provided high heat shorts, with resultant increases in heat transfer and vaporization. Also shown in Figure E-3 are the analytical results for various proportions of liquid heating associated with surface boiling.

To illustrate the range of differences needed to explain the various flights, the data for five flights are shown in Figure E-4. In Figure E-4, curves have been fitted with various allocations of heating between the liquid and ullage. Here again, the range of conditions is bracketed by wetted and dry wall calculations.

Based on the evaluation of oxygen data, a range of values for the amount of liquid heating that should be allocated to surface boiling was observed to range from 75 to 90 percent. The flight data also indicated that for residual oxygen liquid, the most accurate model for definition of equivalent liquid wall contact area was a settled liquid mass.

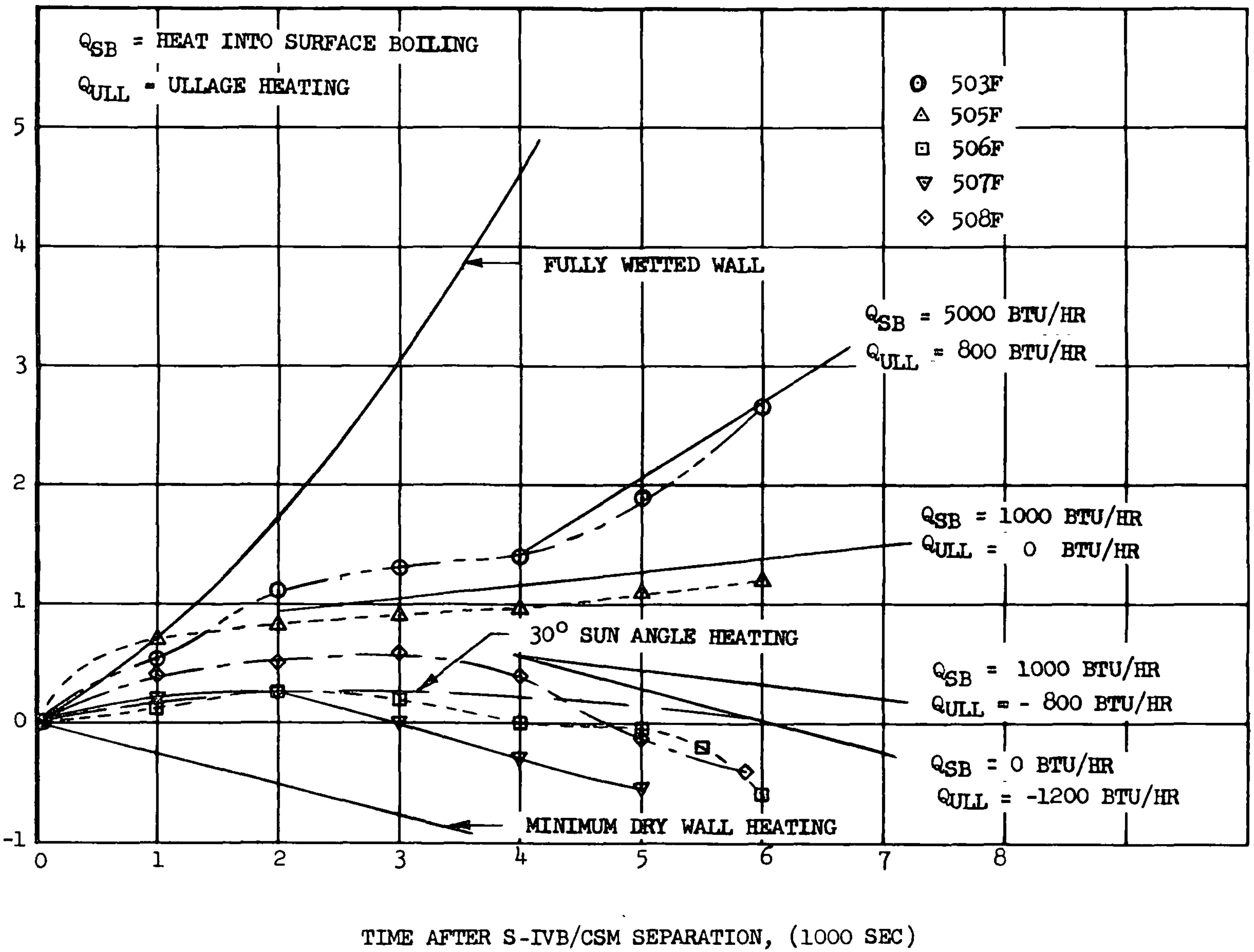
E-2.2 Liquid Hydrogen Evaluation - Segments from 11 flights were examined to obtain data applicable to the residual hydrogen propellant. Six sets of these data were analyzed in depth, using one particularly well-instrumented set to correlate the other data. Emphasis was placed on long steady state periods with a constant low g level applied to the vehicle. These conditions were selected to minimize the influence of the hydrogen tank insulation heat capacity. Periods of no venting, or periods with measured vented gas quantities, were chosen to determine the total mass in the tank. Thus, an accurate accounting of energy could be achieved. Hydrogen tank energy balances were carried out for a number of tank conditions on both the S-IVB and S-IV stages. These helped to determine heat flux to liquid and gas, and percentage of tank wall wetted by liquid during low g operations.

Results of seven energy balances which were applicable to the orbiter are tabulated in Figure E-5 and calculations related to these data are shown in Figure E-6 Figure E-6 presents the observed heat transfer rate as a function of calculated wetted wall area, assuming residuals to be completely settled in the tank base. A key point in Figure E-6 is point number 6. For this set of conditions, an evaluation of heat transfer fully independent of ullage and liquid temperature measurements was obtained. Point 6 corresponds to a steady state, continuous venting period, and represents data from several S-IVB/V flights.



LOX TANK ULLAGE PRESSURE (204 FLIGHT, LOW-G COAST)

FIGURE E-3



LOX TANK ULLAGE PRESSURE CHANGES FOR TRANSLUNAR COASTS OF THE INDICATED FLIGHTS

FIGURE E-4

EVALUATION POINT	g - LEVEL	$\dot{Q}_{BOILOFF}$ BTU/HR	$\dot{Q}_{LIQUID\ HEATING}$ BTU/HR	$\dot{Q}_{ULLAGE}$ BTU/HR	WETTED WALL AREA OF SETTLED RESIDUAL, FT^2	
3	$1X10^{-5}$	72,500	APPROX. 50,000	25,200	955	(38.6%)
4	$1\text{-}3X10^{-4}$	37,000	62,700	25,000	935	(37.8%)
5	-10^{-8}	101,800	33,000	0	425	(17.2%)
6	$4X10^{-5}$	136,000	0	23,500	1550	(68.8%)
7	$-4X10^{-6}$	140,000	10,300	0	160	(6.5%)
11	$2\text{-}7X10^{-4}$	25,000	0	68,200	EQUIVALENT 190	(7.7%)
6A	$4X10^{-5}$	TRANSIENT	57,000	TRANSIENT	1550	(68.8%)

FIGURE E-5

SUMMARY OF HEATING RATES

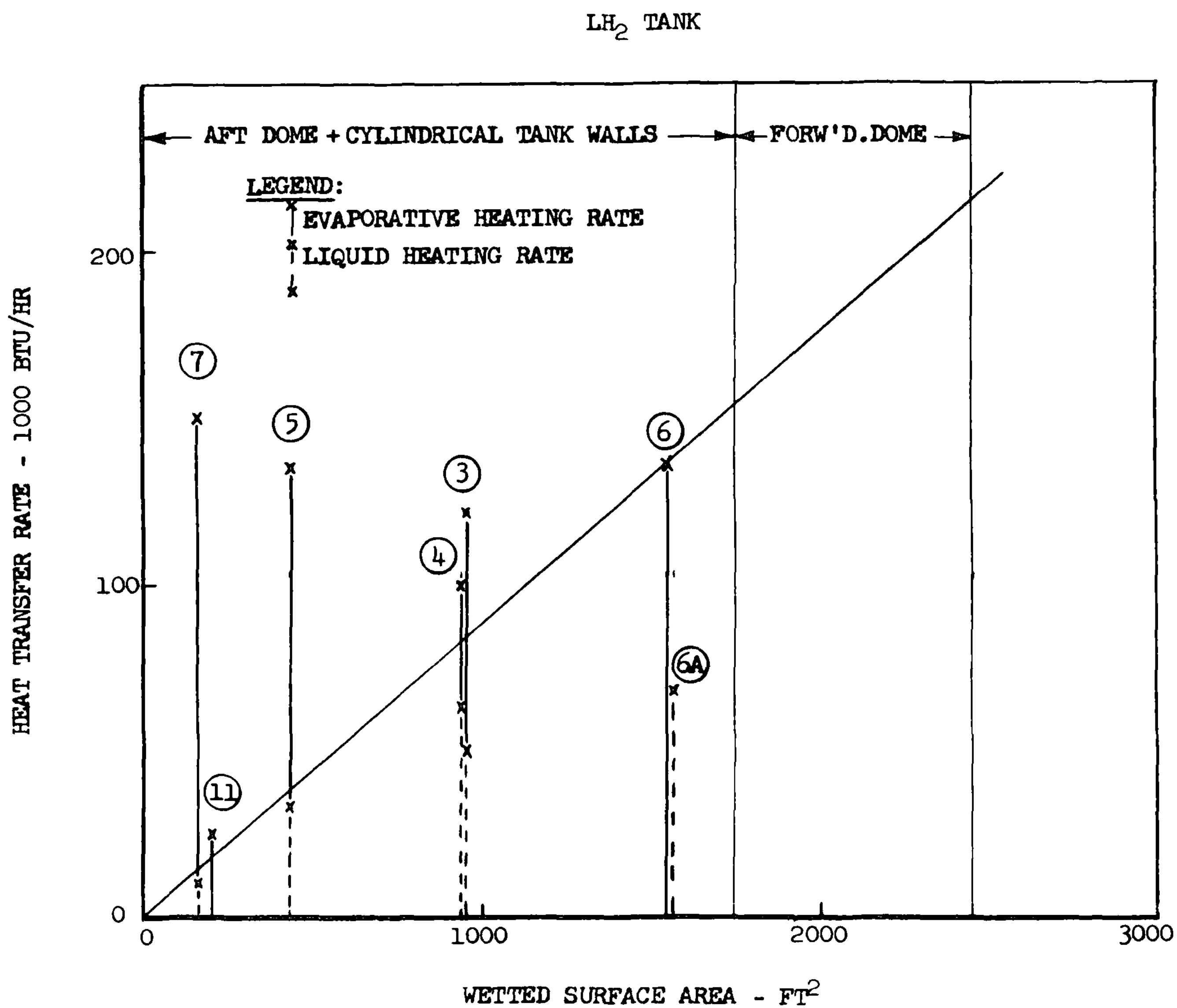


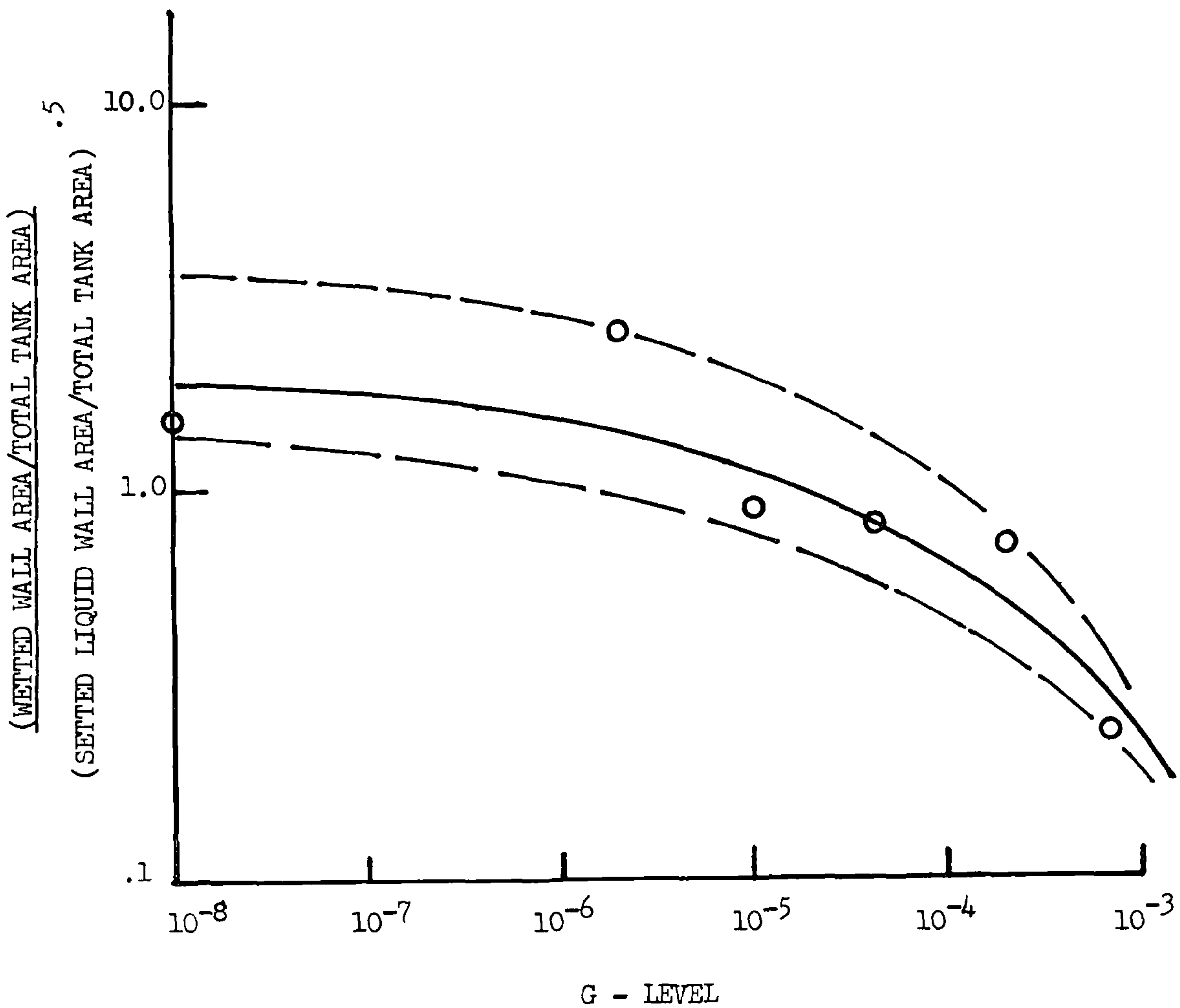
FIGURE E-6

S-IVB, LIQUID AND EVAPORATIVE HEATING RATES VS. WETTED SURFACE AREA OF SETTLED RESIDUAL

Results of an energy balance under the conditions represented by point 6 are reliable, since the hydrogen tank had reached an equilibrium condition with saturated liquid and a stratified ullage gas. Under these conditions, boil-off and ullage heating could be derived directly from measured flow rate and enthalpy. Also, wetted wall area during this process was relatively well known. For these reasons point 6 was considered to be the most reliable point of Figure E-6, and a straight line passing through point 6 was assumed to relate heat transfer rate to wetted surface area.

For the other points in Figure E-6, there was significantly more uncertainty regarding the amount of wall wetted by liquid. However, one major factor neglected in Figure E-6 was the influence of acceleration level on liquid configuration and wetted wall area. From Figure E-5 and E-6 it may be observed that points which are most distant from the linear relationship between liquid/ullage heat input versus wetted wall area, as described by the line through point 6 (for example, points 5 and 7), represent those with the lowest gravity environment. Thus, these points are expected to be the least settled liquid configuration. In order to investigate the influence of gravity, the data from Figure E-5 were correlated using their acceleration levels. The correlation expressed an effective wetted wall fraction as a function of acceleration. The effective wetted wall was defined as that which would be wetted assuming the data for point 6 provided a linear relationship between heat transfer rate and wetted wall area. Each data point in Figure E-6 was translated to the straight line passing through point 6 at a constant heat transfer rate. The wetted area defined by the straight line provided the effective wetted wall area. The original data point provided the settled liquid wetted area. The correlation between the ratio of effective wetted area to settled wetted area and acceleration is shown in Figure E-7.

Application of Saturn data to definition of a liquid hydrogen model indicated that the position of the liquid hydrogen was strongly acceleration-dependent. Correlation of Saturn data with space shuttle data indicated that approximately 65 percent of the hydrogen tank would initially be wetted, even if only small amounts of residual were present. Also, the analysis showed that hydrogen liquid was maintained near saturation conditions. Thus, effectively 100 percent of the liquid heating goes into vaporization.



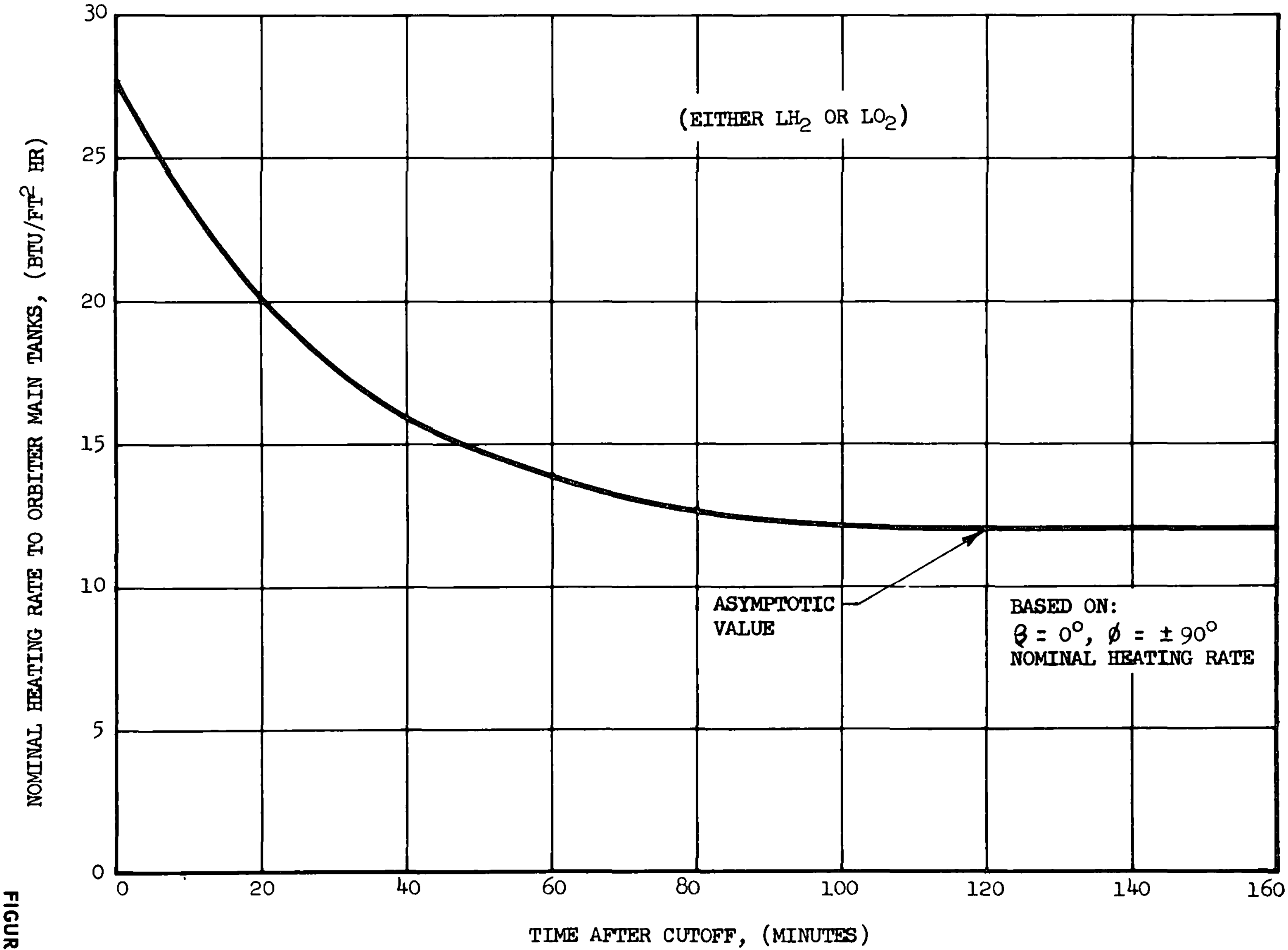
LH2 WALL - WETTING DEPENDENCE ON G - LEVEL

FIGURE E-7

E-3. RESIDUAL AVAILABILITY

E-3.1 Orbiter APS - Using the above models for liquid vaporization, simulated APS missions were analyzed, by coupling liquid boil-off, vapor heating, and APS propellant usage, to determine the usability of liquid residuals. A brief description of analysis and results obtained is presented below for both high and low cross-range orbiters (orbiters A and B, respectively). Heating rates to the residual liquid propellants following main engine shutoff were estimated for orbiter A. Results are shown in Figure E-8. These rates were then applied (in conjunction with the vaporization models developed from the Saturn vehicle study) to obtain the histories for residual liquid remaining presented in Figures E-9 and E-10. Liquid vaporization rates were then computed using these residual histories, and employed in a computer program (which simulates APS mission operation) to determine the amount of residual liquids that could be used by the APS. A typical computer mission simulation is shown in Figure E-11. Results from these computer mission simulations are summarized in Figure E-12 for three distinct maneuver velocity allocations. (The liquid vaporization rates used for the orbiter A analysis were also assumed valid for Vehicle B.) As seen in Figure E-12, only a small percentage of the initial LOX residual is usable by the APS. This is because LOX vaporization rate is relatively large, and most of the oxygen vapor is vented before major vehicle velocity changes are performed. The usable residual liquid weights presented in Figure E-12 were applied to the propellant weight requirements tabulated in the detailed subsystem weight breakdowns of Appendix F.

E-3.2 Booster APS - Utilization of main engine tank propellant residuals was also considered for the booster APS. Simulated mission evaluations for the booster APS clearly showed that ample residual propellant vapors were available in the main engine tanks to satisfy the entire booster mission without propellant resupply. An example of the vehicle B booster mission simulation is shown in Figure E-13. Based on these results, no additional analyses of booster residual liquid vaporization were undertaken in this study.



NOMINAL HEATING RATE TO WET WALL AFTER ORBITER MAIN ENGINE CUTOFF

FIGURE E-8

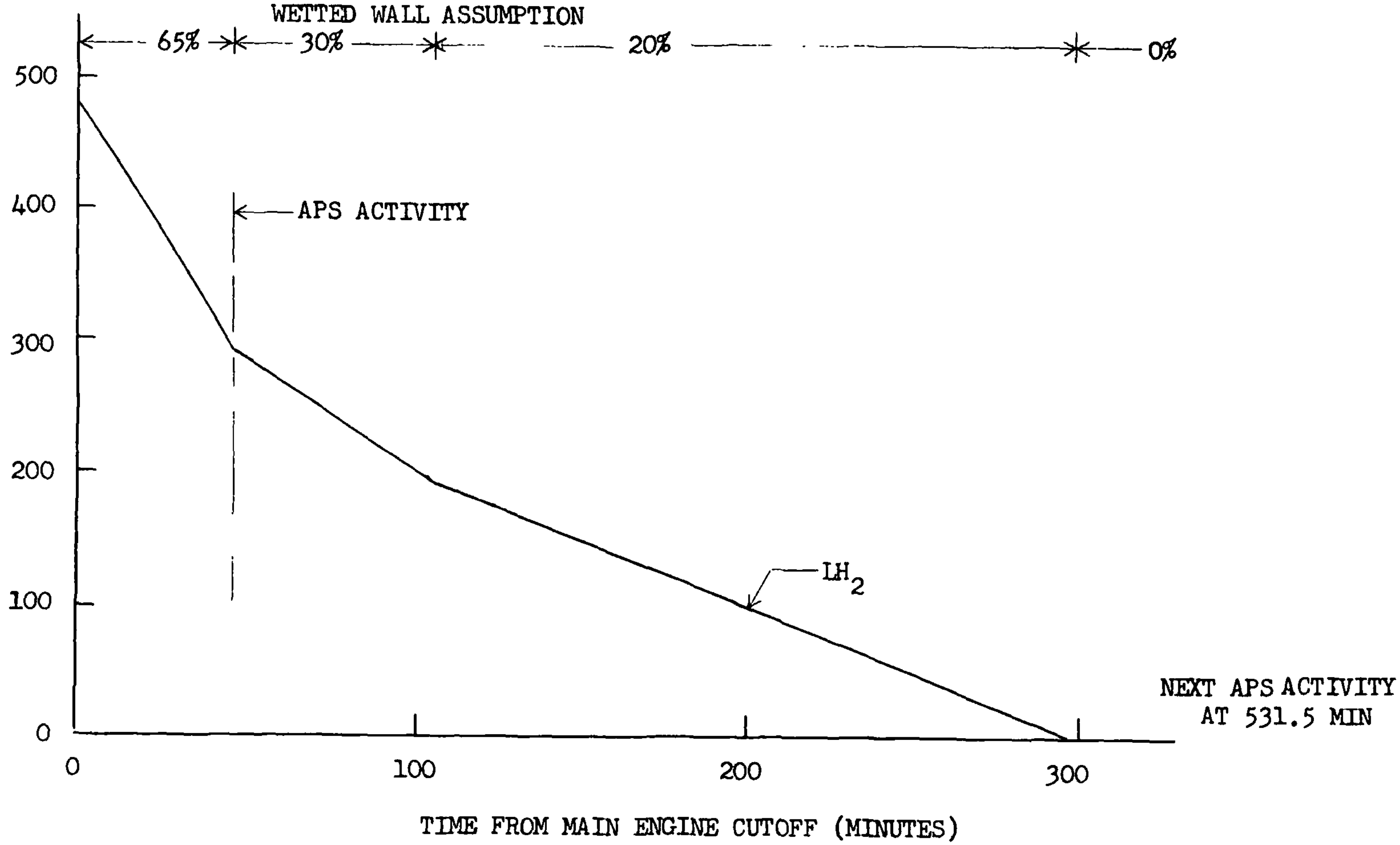
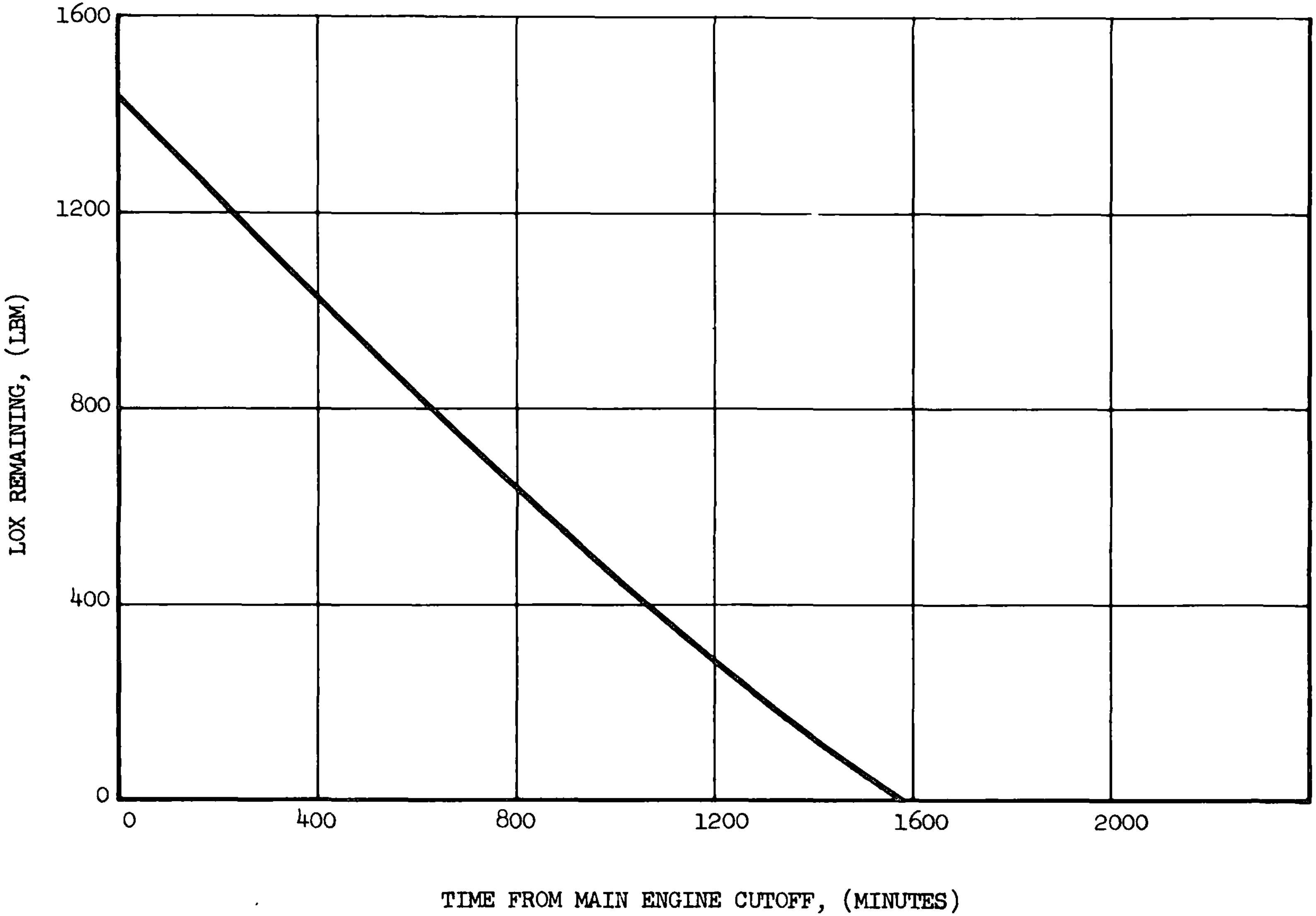


FIGURE E-9

ORBITER A LH_2 TIME HISTORY

SETTLED-LIQUID-HEATING



ORBITER A LOX TIME HISTORY ALL MISSIONS

FIGURE E-10

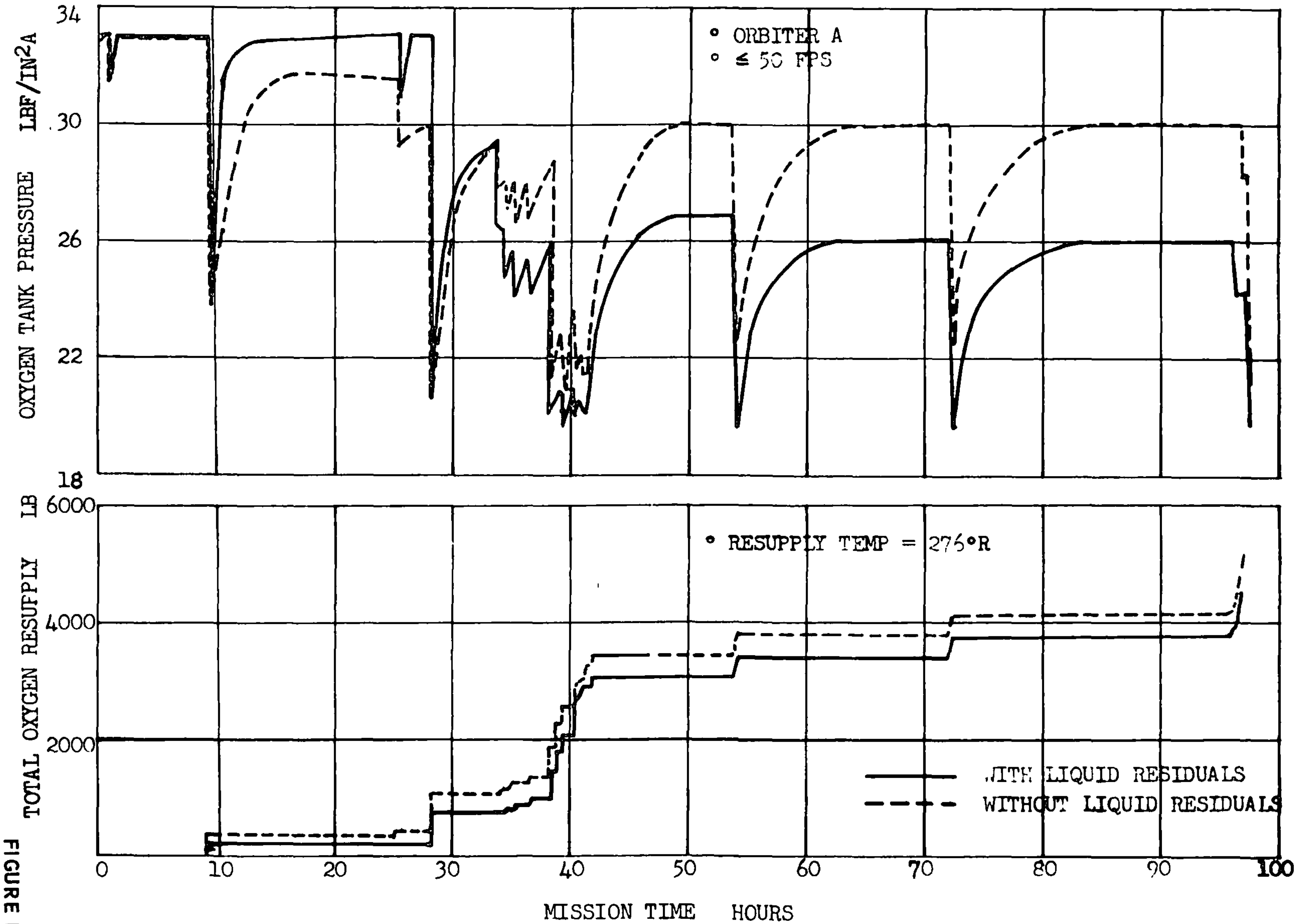


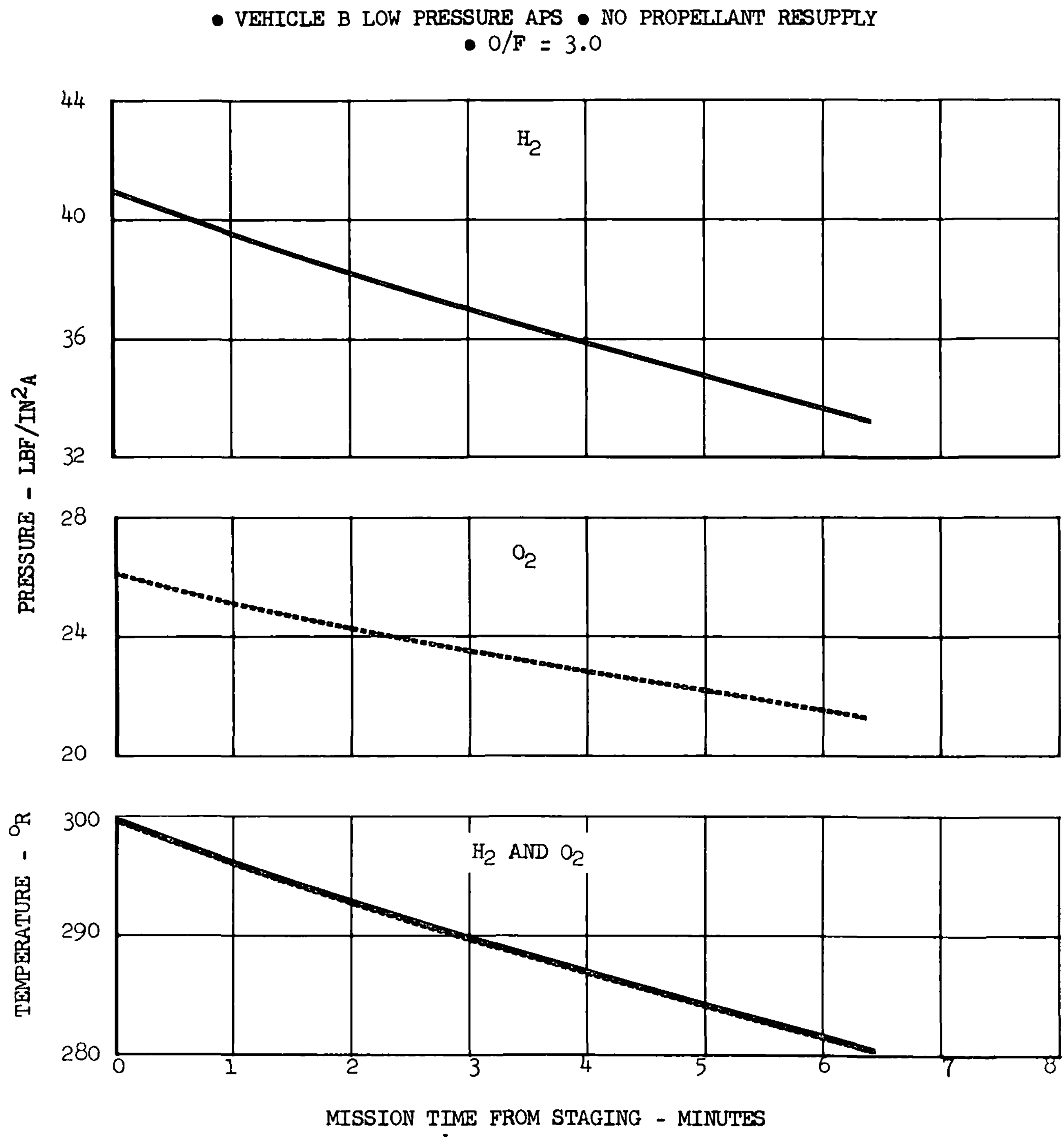
FIGURE E-11

EFFECT OF BOOST RESIDUALS ON APS RESUPPLY REQUIREMENTS

VEHICLE	PROPELLANT	LIQUID RESIDUAL (LB)	APS USABLE		
			≤ 10 FPS	≤ 50 FPS	ALL MANEUVERS
ORBITER A	O_2	1,827	262	365	643
	H_2	510	125	163	362
ORBITER B	O_2	670	75	118	139
	H_2	31	0	27	31

FIGURE -12

APS UTILIZATION OF MAIN ENGINE TANK LIQUID RESIDUALS



BOOSTER TANK PRESSURE AND TEMPERATURE HISTORIES

FIGURE E-13

APPENDIX F

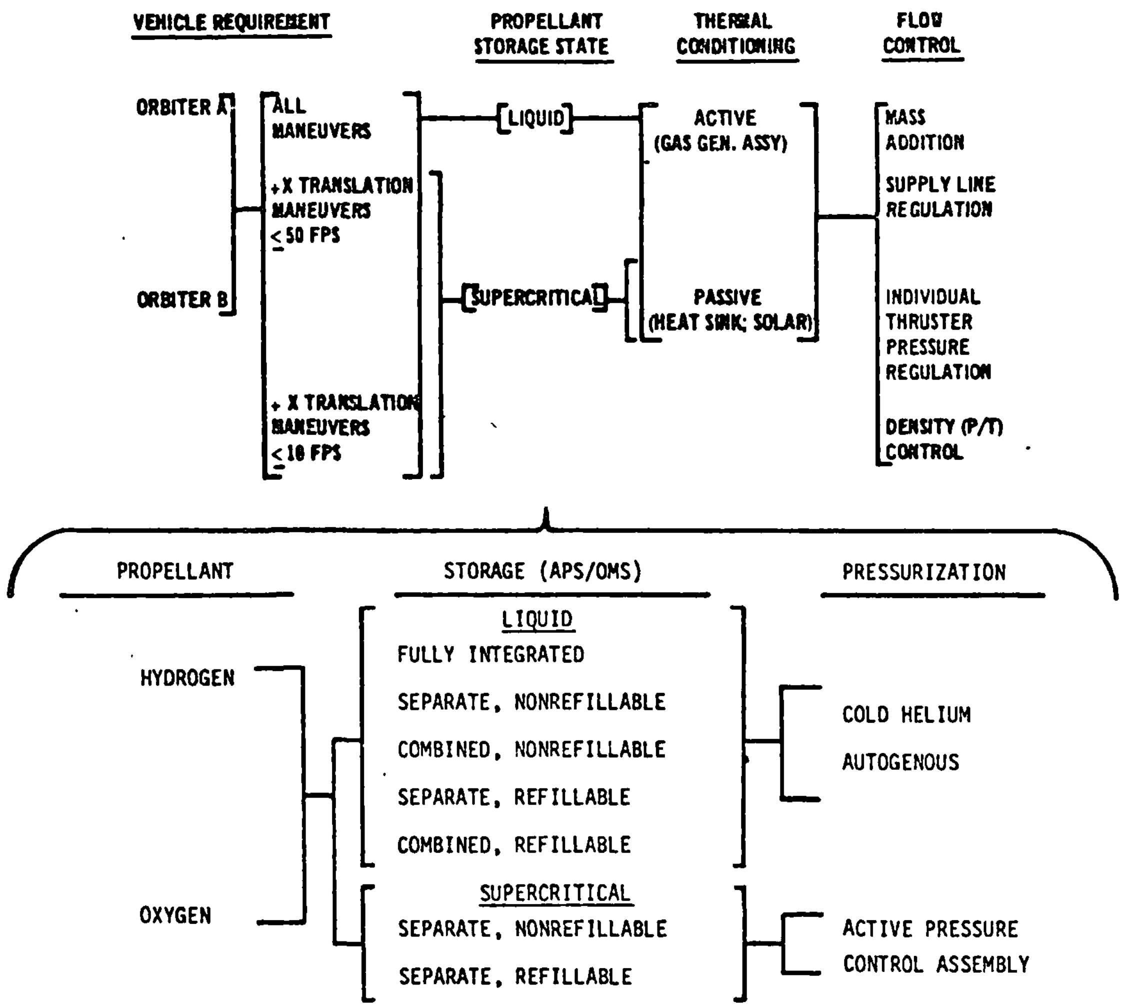
APS Design and Weight Sensitivity

F-1. INTRODUCTION

Design and weight sensitivity analyses were conducted on 68 candidate low pressure APS concepts. All but four of these pertained to orbiter vehicles. Various approaches to vehicle requirements, propellant storage state, thermal conditioning, and flow control devices were investigated as required for each of two booster and orbiter elements. Space shuttle configurations considered were low (shuttle A) and high (shuttle B) crossrange, 2-stage, fully reusable space vehicles. Various velocity allocations between APS and an orbit maneuvering subsystem (OMS) were considered. These were:

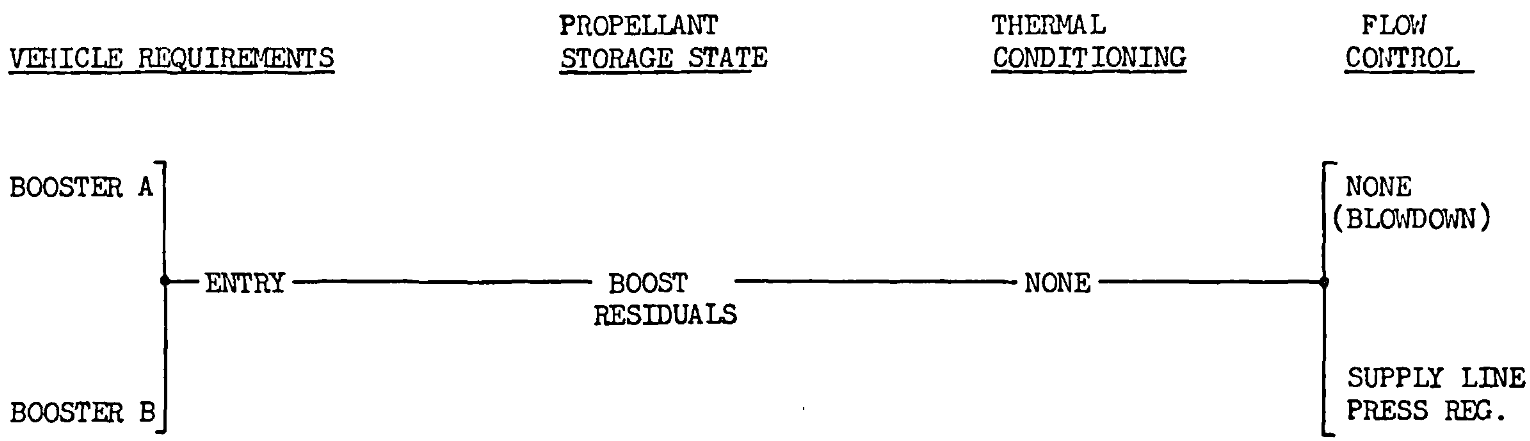
(1) APS provides all attitude control and +x translational maneuvers $\leq$ 10 ft/sec
(2) APS provides all attitude control and +x translational maneuvers $\leq$ 50 ft/sec
(3) APS provides all attitude control and all translational maneuvers.

The booster APS provides for all attitude control maneuvers with residual propellants. Therefore, it requires only a propellant distribution network and control engines. Matrices of candidate APS concepts for orbiters and boosters are presented in Figures F-1 and F-2, respectively. Detailed subsystem schematics are presented below for the most attractive concepts. The schematic legend is shown in Figure F-3.



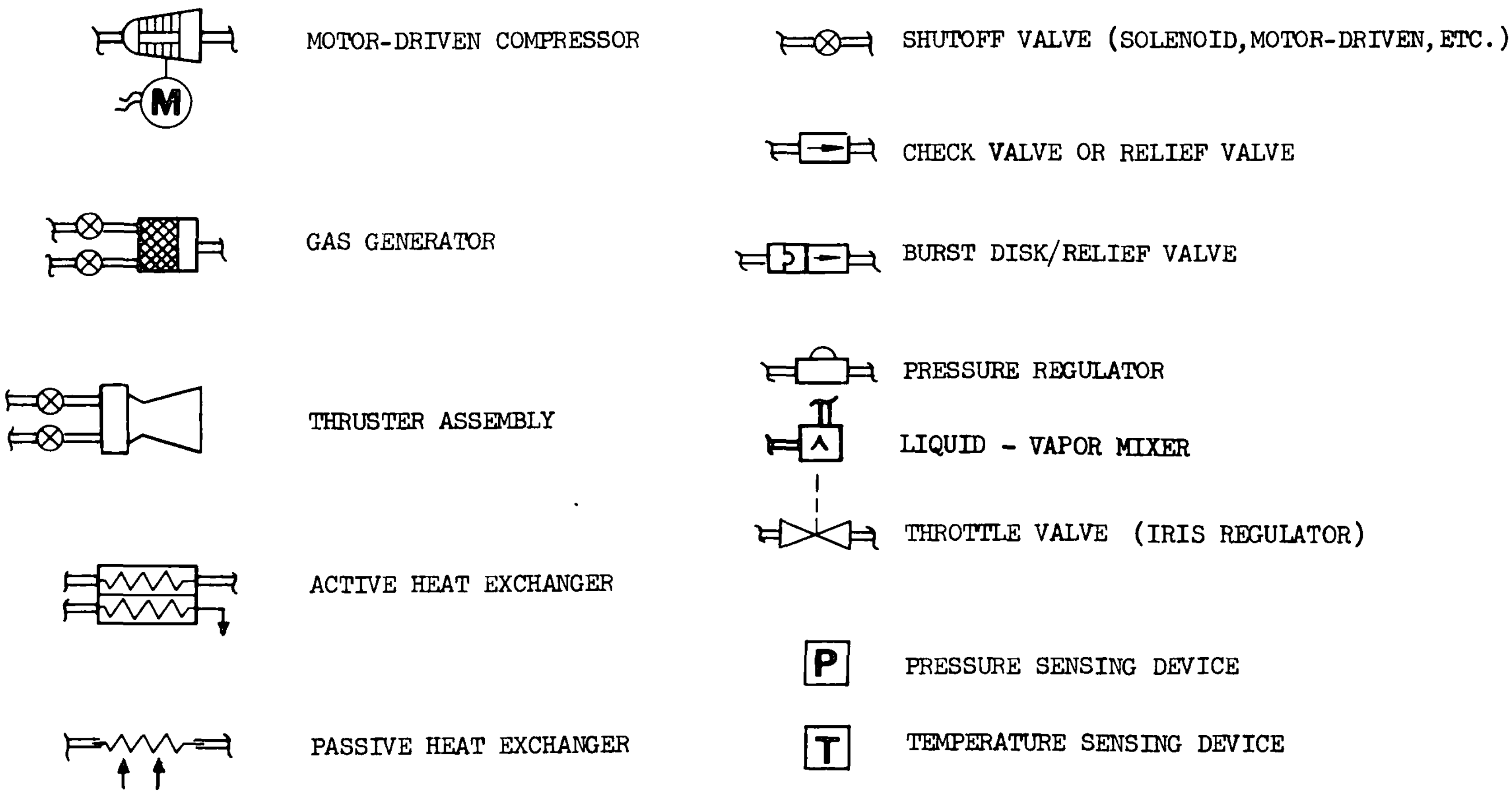
ORBITER CONCEPT MATRIX

FIGURE F-1



BOOSTER CONCEPT MATRIX

FIGURE F-2



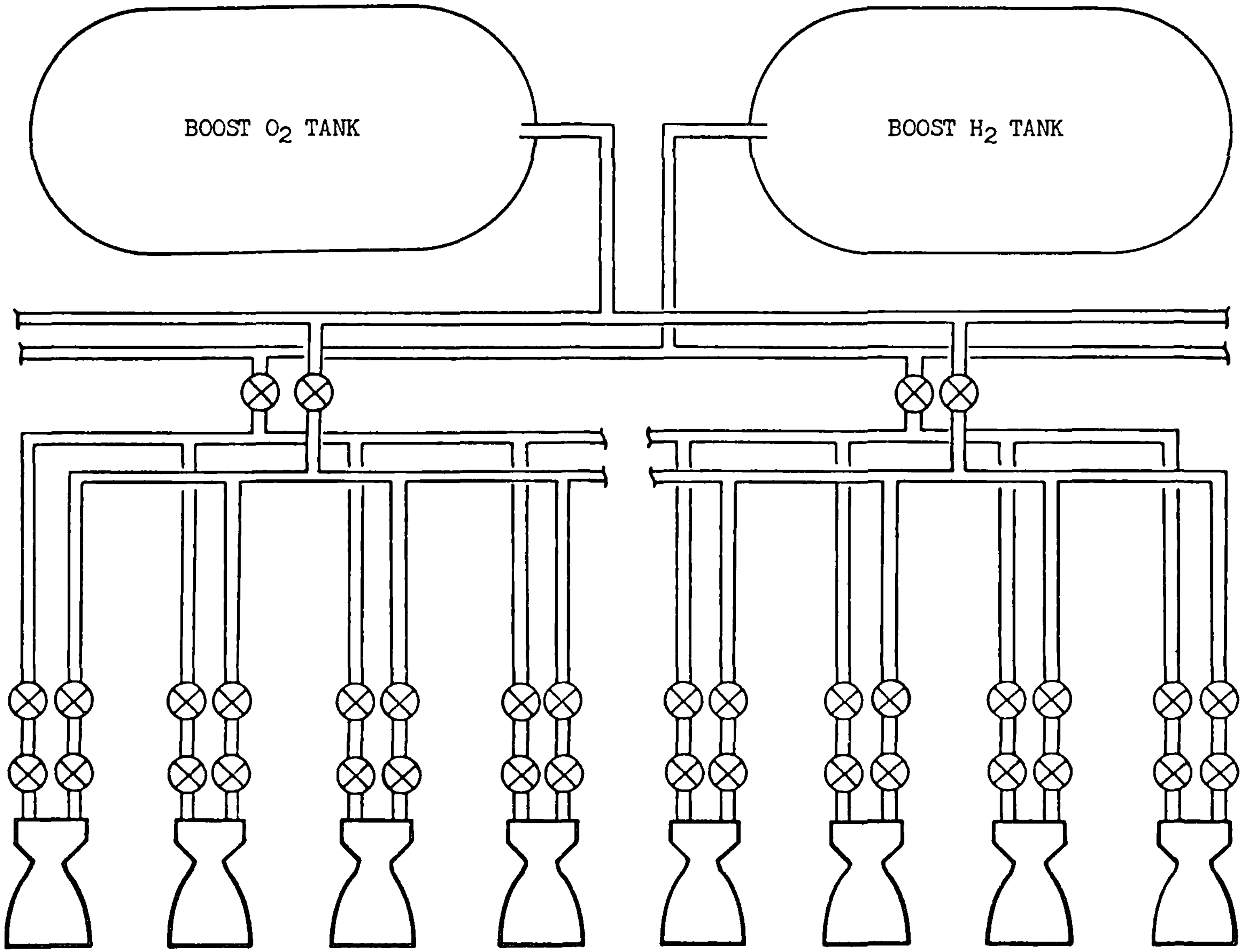
SCHEMATIC LEGEND

FIGURE F-3

F-2. CONCEPT DESCRIPTION - BOOSTER

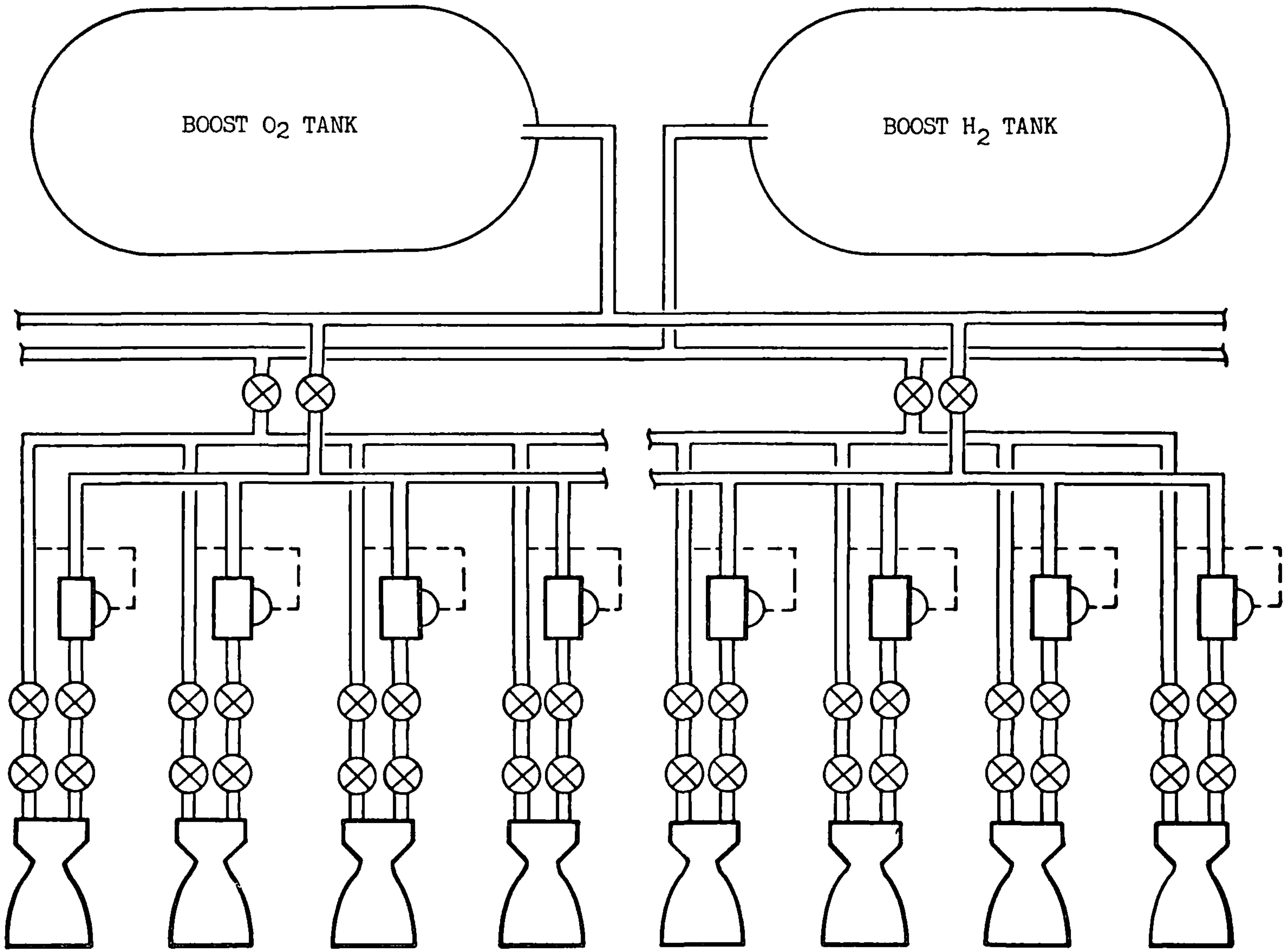
Booster APS mission performance requirements are achieved utilizing only main engine tank propellant residuals thus eliminating the need of auxiliary tankage, and thermal conditioners. Hence, only two candidate auxiliary propulsion subsystems were evaluated for boosters A and B. These were: (1) a simple gas blowdown concept shown schematically in Figure F-4 which requires only distribution lines, isolation valves, and engine assemblies, and (2) a differential pressure regulation control concept shown in Figure F-5, requiring a regulator for each engine assembly in addition to the above mentioned components. Performance requirements were met with a booster A configuration using eighteen 2600 lb thrust engines and booster B configuration of sixteen 2000 lb thrust engines. Subsystem concept design optima are summarized in Figure F-6. For all cases considered, engine mixture ratios optimized at 4.0 and nozzle expansion ratios at 2.0. Maximum distribution line diameters varied from 6.9 to 8.3 in for fuel lines and from 7.2 to 8.7 in for oxidizer lines.

FIGURE F-4



BOOSTER SUBSYSTEM SCHEMATIC

. BLOWDOWN



BOOSTER SUBSYSTEM SCHEMATIC
. ΔP REGULATION

FIGURE F-5

	BOOSTER A		BOOSTER B	
	BLOWDOWN	REGULATED	BLOWDOWN	REGULATED
SYSTEM OPTIMUMS				
P_c	21.9	18.75	14.0	11.0
MR	4.0	4.0	4.0	4.0
ϵ	2.0	2.0	2.0	2.0
MAX LINE DIAMETER				
H_2	7.1	7.9	6.9	8.3
O_2	7.5	8.3	7.2	8.7
SYSTEM WEIGHT				
PROPELLANT SYSTEM	← NO APS PROPELLANT REQUIRED, BOOST SYSTEM RESIDUAL GAS UTILIZED →			
ISOLATION VALVES, REGS	759	1246	672	1116
LINES	625	771	806	1161
ENGINES	1588	1773	1611	1941
TOTAL SYSTEM WEIGHTS	2972	3690	3089	4210

FIGURE F-6

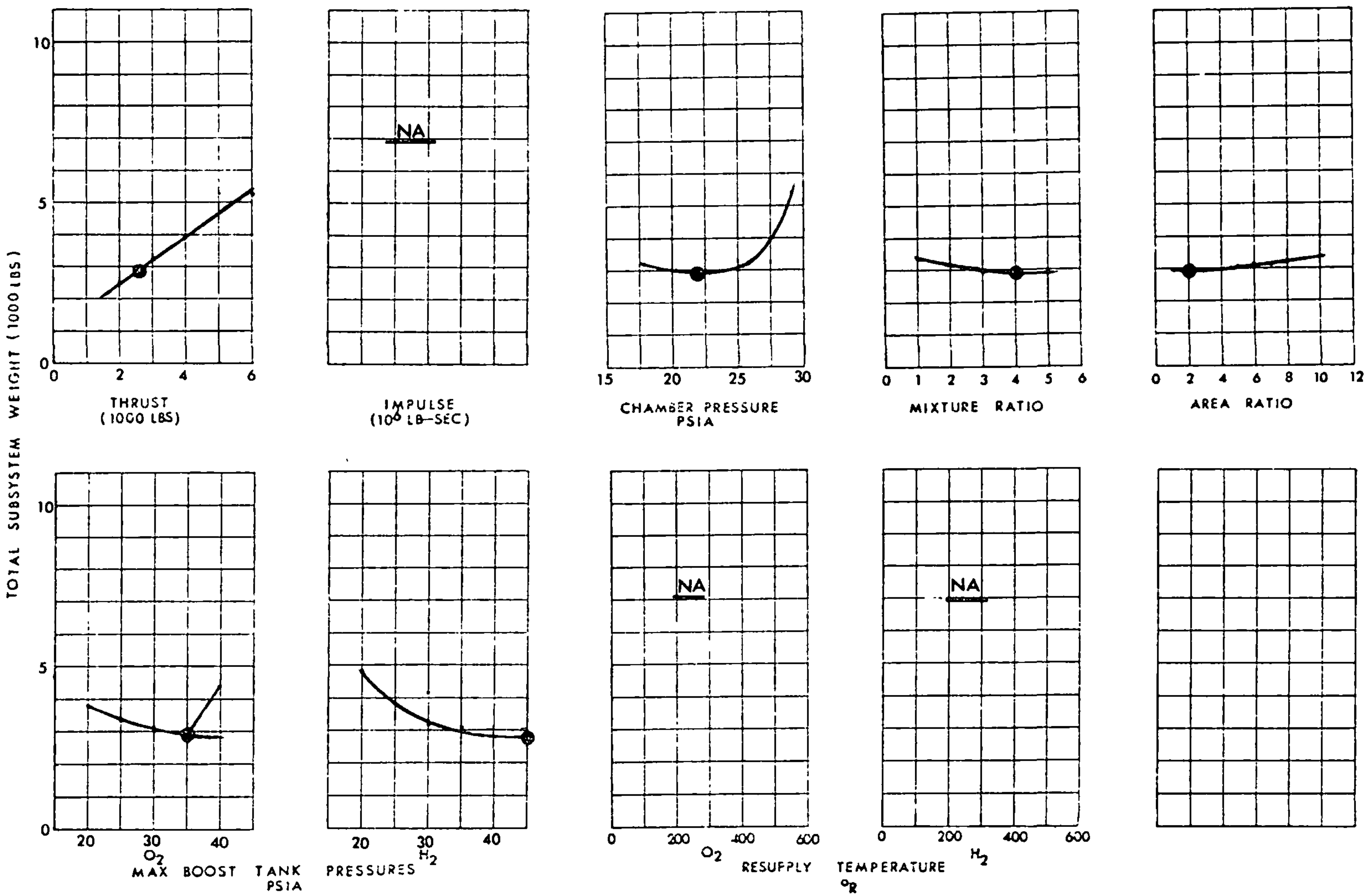
BOOSTER APS CONCEPT COMPARISON

F-3. SUBSYSTEM WEIGHT - BOOSTER

Distribution and engine assemblies were sized to provide the desired thrust level at end of mission when booster tank pressures and temperatures had decayed to their lowest values. In addition, the propellant distribution network (lines and isolation valves) was sized to provide minimum subsystem weight by trading-off the line weight against engine weight. An engine injector and valve pressure differential of 2 lbf/in^2a was estimated to be the minimum allowable for good mixing and stable operation. Component and total subsystem weights are tabulated in Figure F-6 for each concept and booster vehicle. The simple blowdown concept resulted in the lightest subsystem weight, specifically 2972 and 3089 lb for boosters A and B, respectively.

F-4. DESIGN AND WEIGHT SENSITIVITIES - BOOSTER

Linear sensitivities to thrust, engine chamber pressure, mixture ratio, nozzle expansion ratio, and maximum boost tank pressure were evaluated for both booster APS concepts. Linear sensitivities were evaluated by varying a single design parameter while maintaining all other design parameters at their design point. Sensitivities and design point optima are shown in Figures F-7 through F-10. Sensitivity to total impulse and resupply propellant conditioning temperatures are not shown since sufficient residual propellant was available to meet all performance requirements. In the analysis, main engine tank weight increases were assessed against APS weight for cases where boost tank pressure exceeded design pressure. This is shown by the linearly increasing solid lines to the right of the design point in the curves for APS weight sensitivity to main engine tank pressure shown in Figures F-7 through F-10. In general, subsystem weight was found to be very sensitive to thrust level, chamber pressure, and main engine tank pressures, but insensitive to engine mixture ratio and nozzle expansion ratios.



LOW PRESSURE APS

BOOSTER A

BLOWDOWN

FIGURE F-7

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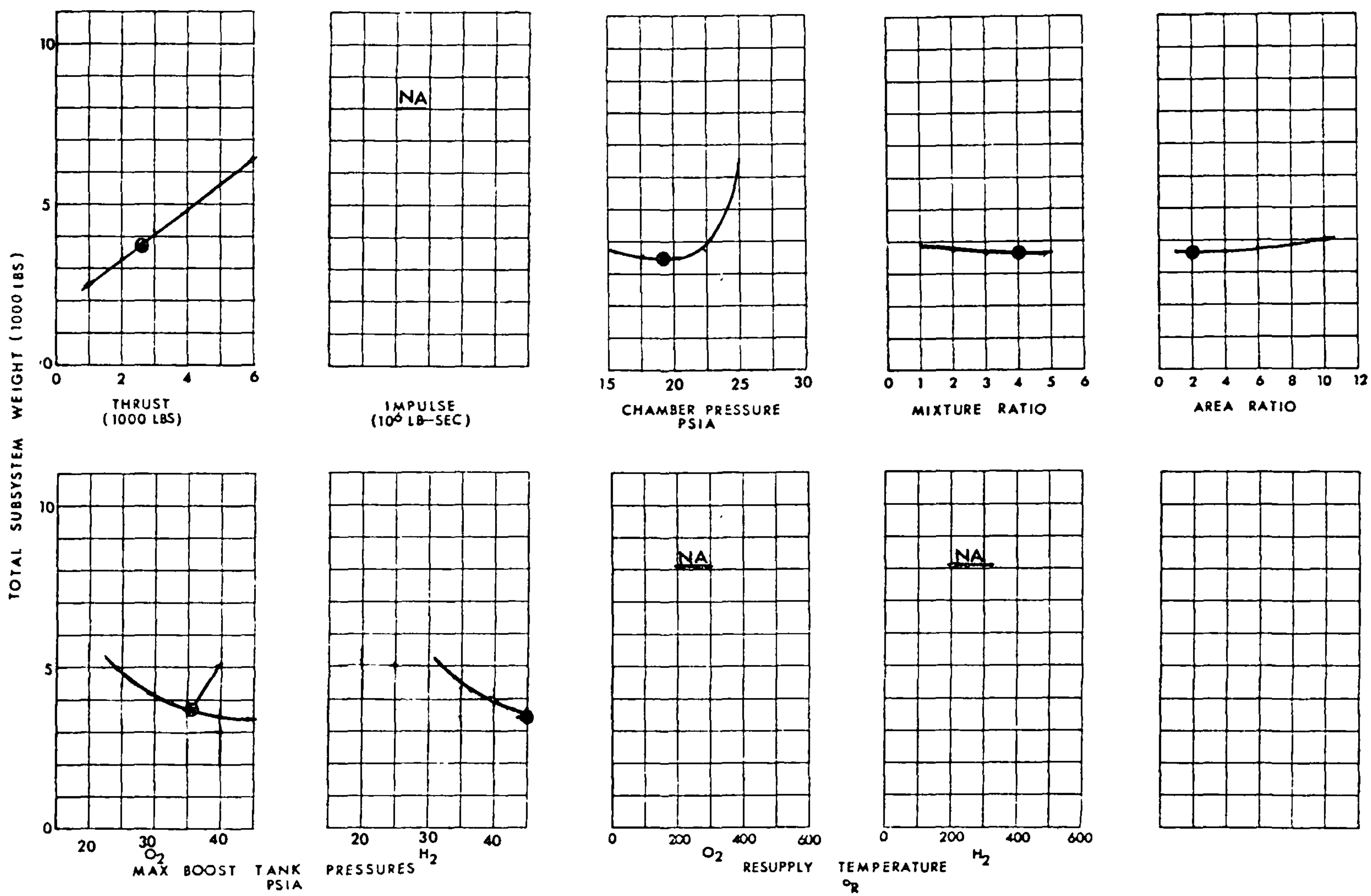
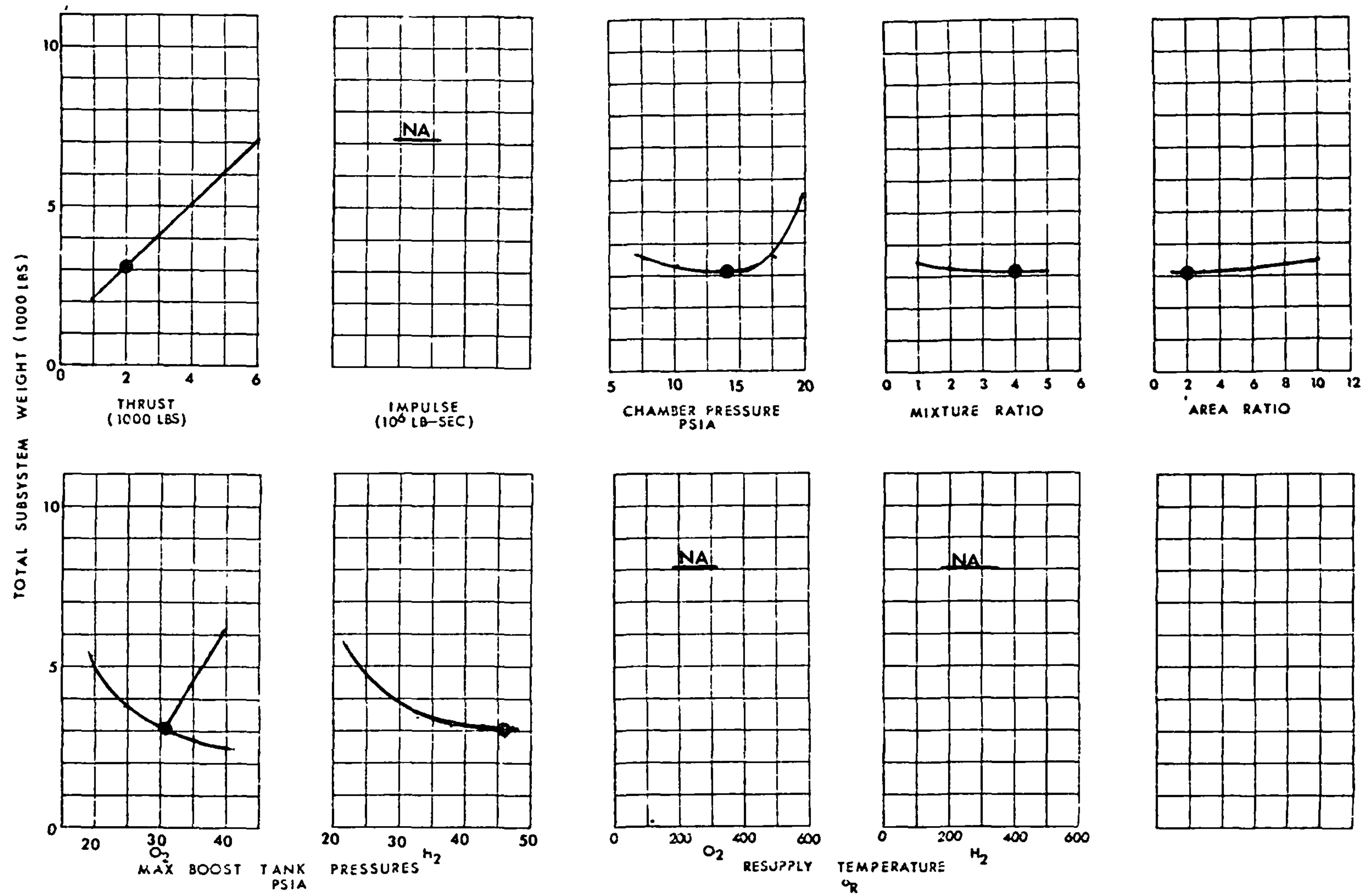


LOW PRESSURE APS

BOOSTER A

REGULATED

FIGURE F-8

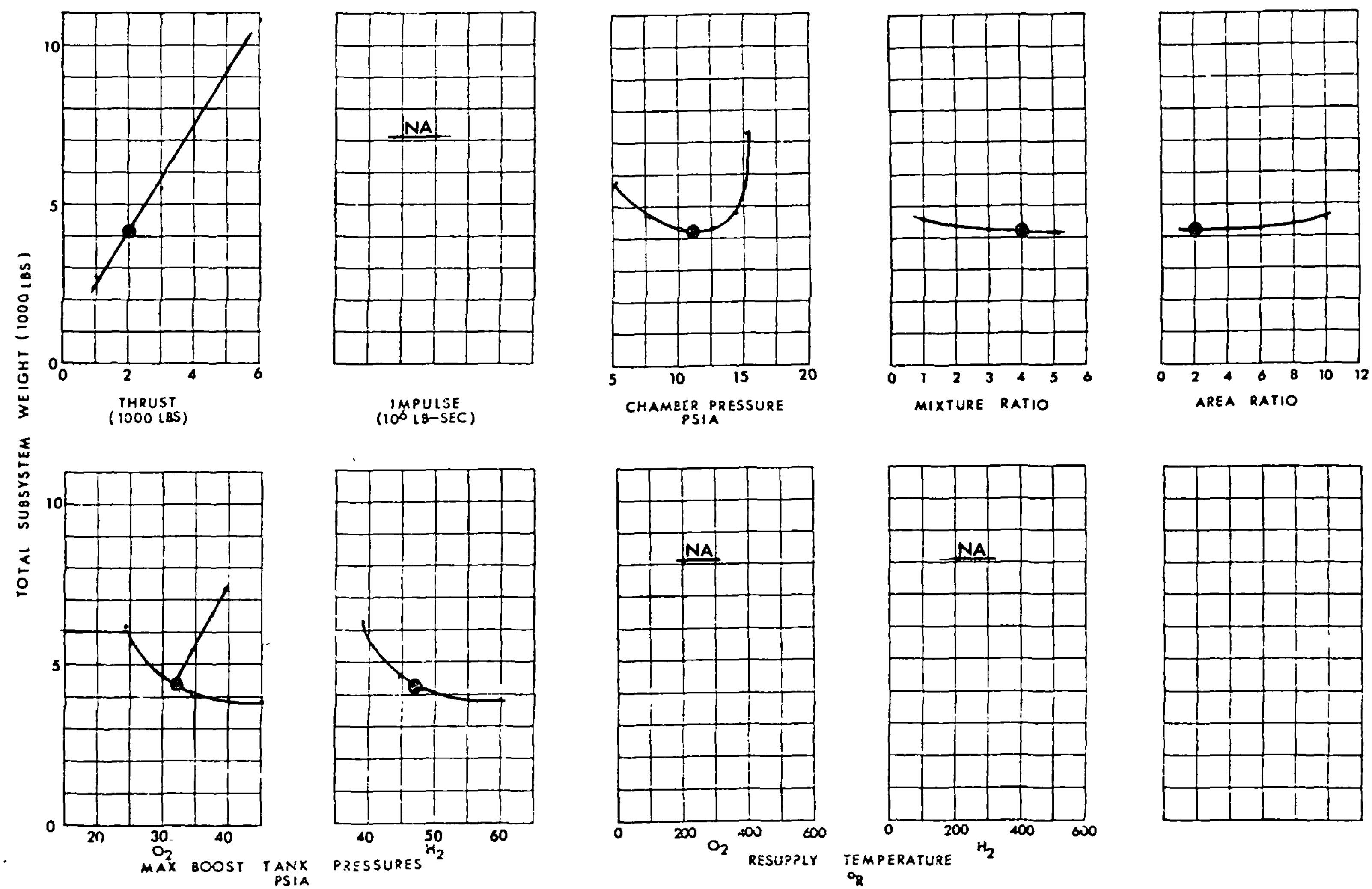


LOW PRESSURE APS

BOOSTER B

BLOWDOWN

FIGURE F-9



LOW PRESSURE APS

BOOSTER B

REGULATED

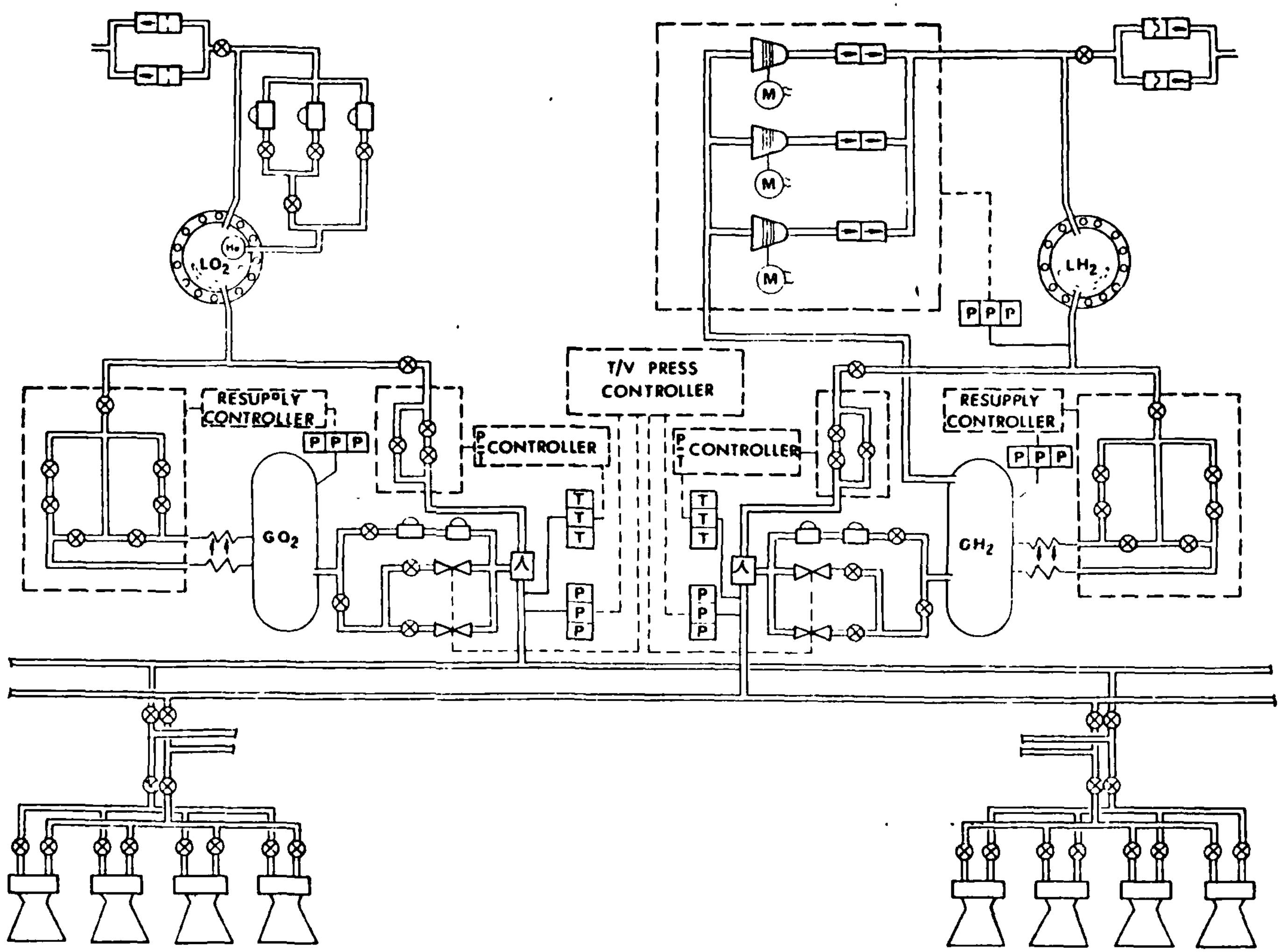
FIGURE F-10

F-5. CONCEPT DESCRIPTION - ORBITER

The design concepts presented in Section F-1 were evaluated at the three predefined velocity allocations for both Orbiter A and B as shown in the concept matrix of Figure F-1. Comparisons of the flow control concepts, as defined in the body of this report, indicated the constant density liquid/vapor mixer approach to be the most attractive based primarily on weight and mission flexibility considerations, and hence subsystem schematics are presented only for this control concept.

The schematics shown in Figures F-11 through F-14 are for liquid storage/passive conditioning, liquid storage/active conditioning, supercritical storage/passive conditioning, and supercritical storage/active conditioning, respectively. All subsystem approaches include the following assemblies: propellant storage tanks with associated pressurization subassemblies; heat exchangers with associated valves and controls; main engine tanks, which serve as gas accumulators; flow control devices; distribution network with associated isolation valves; engines with associated valves, injector, and nozzle. Mission performance requirements were met with an orbiter A configuration utilizing thirty-two 500 lb thrust engines and an orbiter B configuration of twenty-eight 1000 lb thrust engines.

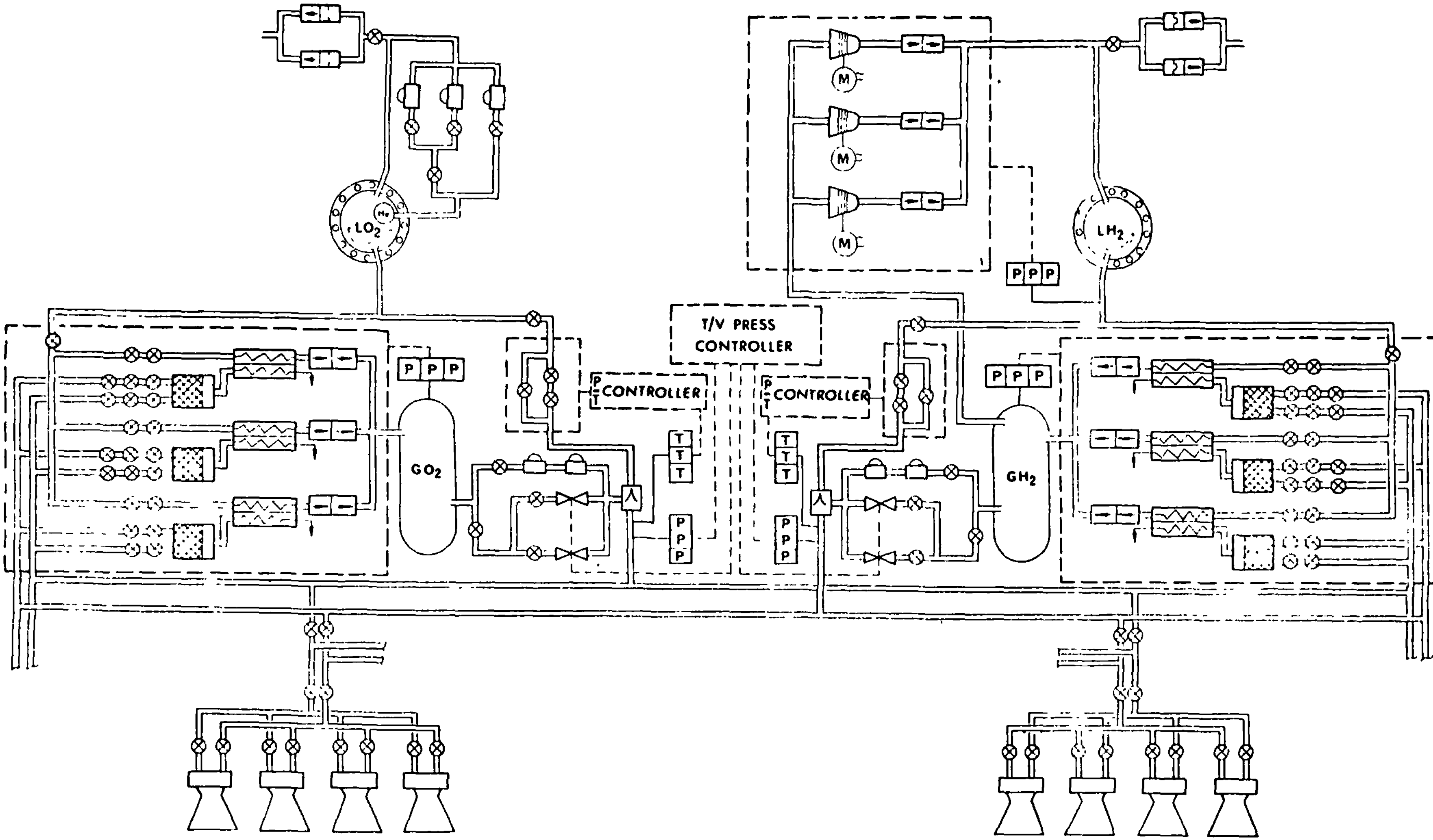
Subsystem design point summaries are tabulated in Figure F-15 (orbiter A) and Figure F-16 (orbiter B) for the constant density control concept. APS total impulse and conditioning temperature requirements, subsystem optima, and subsystem specific impulse and total weight are given for each of the subsystem concepts investigated. Over the entire concept matrix (including both vehicles) optimum engine chamber pressure only varied from 13.3 to 13.9 lbf/in^2a and the optimum mixture ratio remained constant at 3.0. For the 10 ft/sec velocity case, the optimum nozzle expansion ratio was 8.0, while for higher velocity maneuvers the optimum was 10.0. It should be noted that the subsystem specific impulse is based on the combined impulse provided by the APS tanked propellant and the main engine tank residuals.



ORBITER SUBSYSTEM SCHEMATIC

- o LIQUID STORAGE
- o PASSIVE CONDITIONING
- o P/T CONTROL

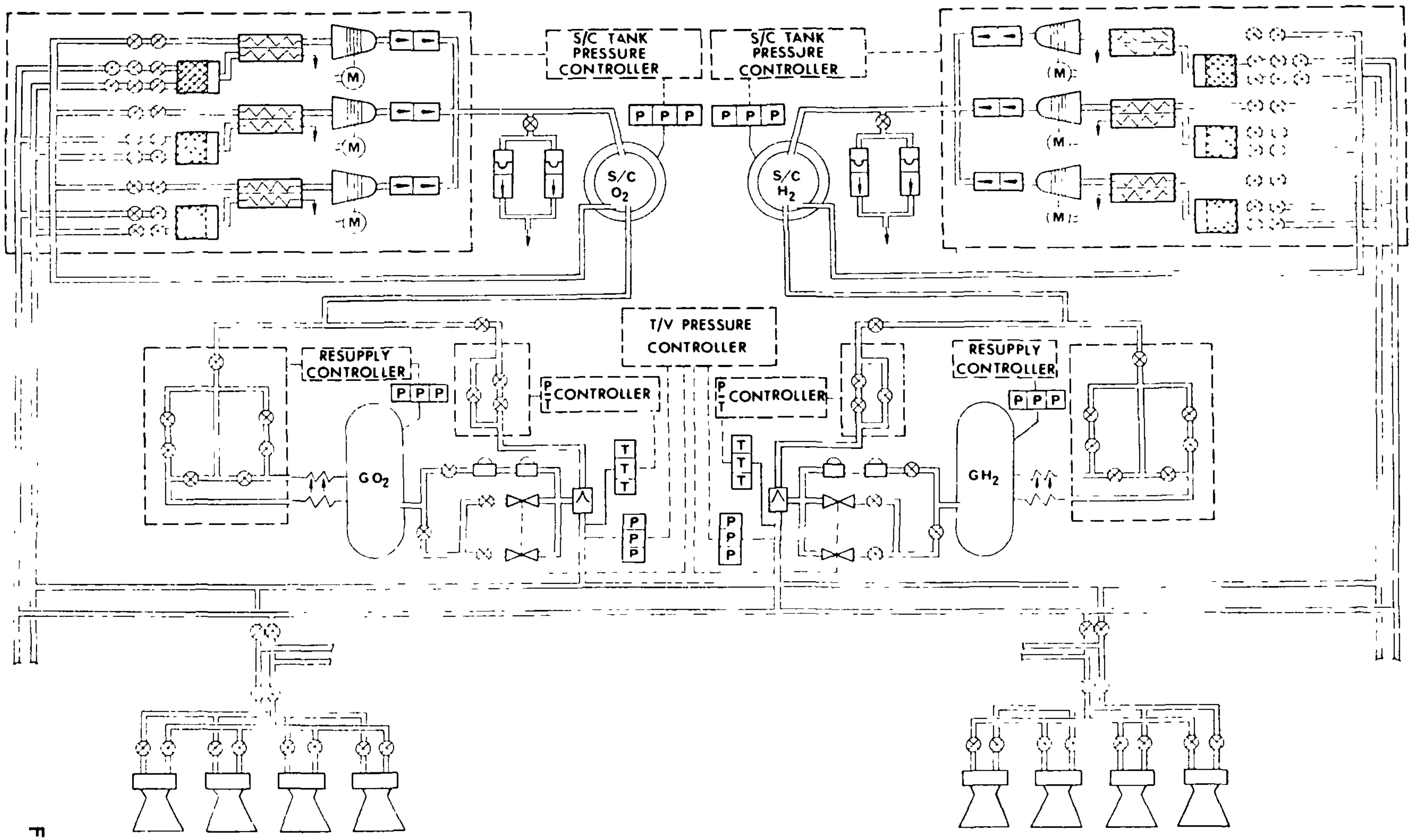
FIGURE F-11



ORBITER SUBSYSTEM SCHEMATIC

o LIQUID STORAGE
o ACTIVE CONDITIONING
o P/T CONTROL

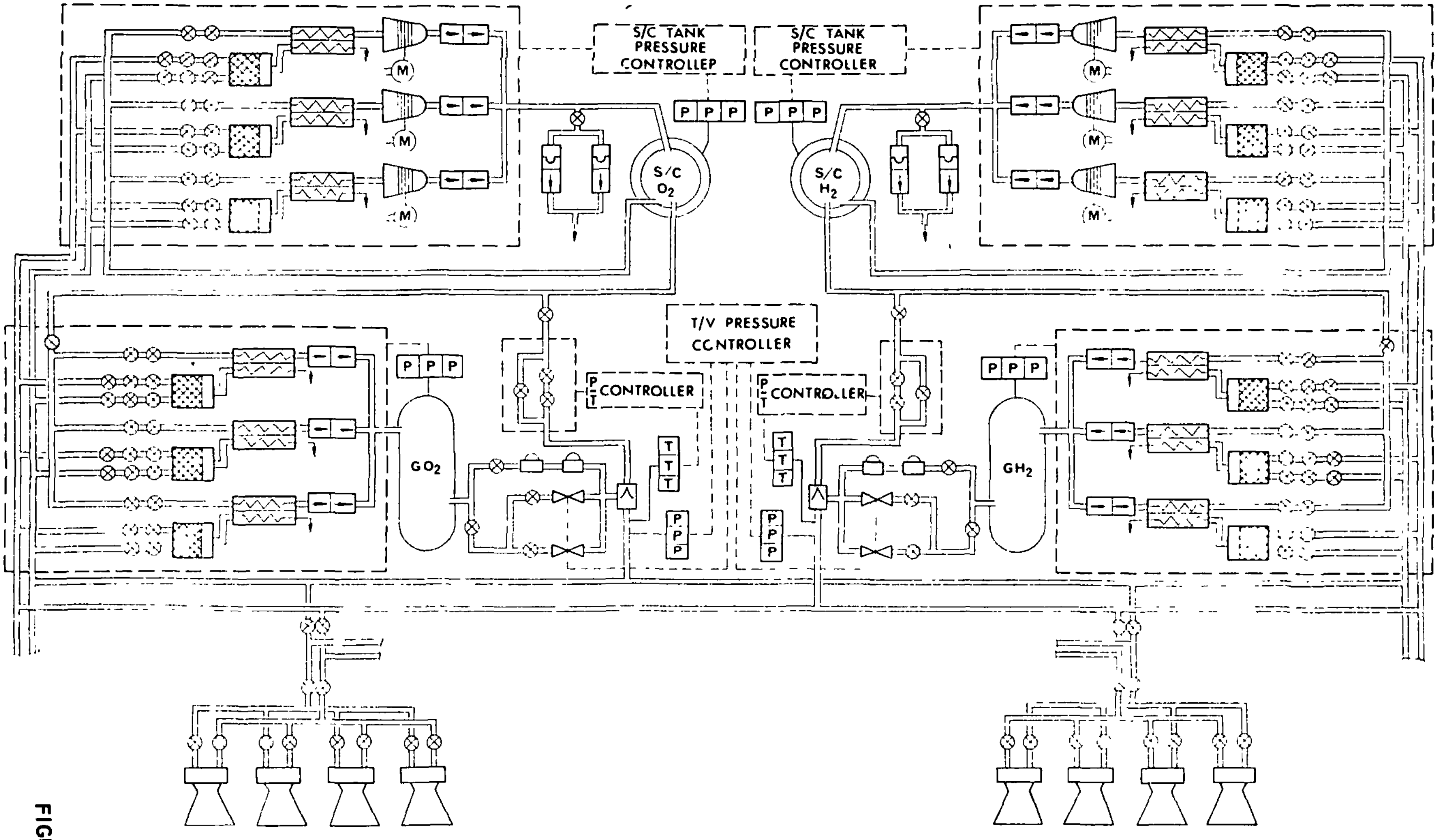
FIGURE F-12



ORBITER SUBSYSTEM SCHEMATIC

- o SUPERCRITICAL STORAGE
- o PASSIVE CONDITIONING
- o P/T CONTROL

FIGURE F-13



ORBITER SUBSYSTEM SCHEMATIC

- SUPERCRITICAL STORAGE
- ACTIVE CONDITIONING
- P/T CONTROL

FIGURE F-14

	TOTAL IMPULSE	SYSTEM OPTIMUMS			SYSTEM	CONDITIONING TEMP		TOTAL SYS WEIGHT
		P_C	ϵ	MR	I_{SP}	H_2	O_2	
$\leq$ 10 FT/SEC								
LIQUID ACTIVE	1.20 (10^6)	13.3	8	3	456	SAT'D LIQUID	SAT'D VAPOR	6755
LIQUID PASSIVE					465			6672
S/C ACTIVE					441			7728
S/C PASSIVE	1.201 (10^6)	13.3	8	3	441	SAT'D LIQUID	SAT'D VAPOR	7421
$\leq$ 50 FT/SEC								
LIQUID ACTIVE	2.698 (10^6)	13.4	10	3	391	100^oR	265^oR	11735
LIQUID PASSIVE					405			11632
S/C ACTIVE					395			12190
S/C PASSIVE	2.698 (10^6)	13.4	10	3	397	100^oR	265^oR	11762
ALL MANEUVERS								
LIQUID ACTIVE	13.426 (10^6)	13.4	10	3	360	236^oR	328^oR	44960

SUBSYSTEM DESIGN POINT SUMMARY

o ORBITER A

o P/T CONTROL

FIGURE F-15

	TOTAL IMPULSE	SUBSYSTEM OPTIMUMS			SYSTEM I_{SP}	CONDITIONING TEMP		TOTAL SYS WEIGHT
		P_C	ε	MR		H_2	O_2	
≤ 10 FT/SEC								
LIQUID ACTIVE	1.523 (10^6)	13.8	8	3	409	184	190	8927
LIQUID PASSIVE					430			8734
S/C ACTIVE					418			10580
S/C PASSIVE	1.523 (10^6)	13.8	8	3	418	184	190	10070
≤ 50 FT/SEC								
LIQUID ACTIVE	3.311 (10^6)	13.9	10	3	375	226	190	14550
LIQUID PASSIVE					395			14543
S/C ACTIVE					380			15818
S/C PASSIVE	3.311 (10^6)	13.9	10	3	389	226	190	15154
ALL MANEUVERS								
LIQUID ACTIVE	15.780 (10^6)	13.9	10	3	356	288	266	52622

SUBSYSTEM DESIGN POINT SUMMARY

o ORBITER B

o P/T CONTROL

FIGURE F-16

F-6. SUBSYSTEM WEIGHT - ORBITER

The orbiter APS was sized in essentially the same manner as described in Section F-3 for the booster. Distribution and engine assemblies were sized based on the lowest values of pressure and temperature occurring during the mission duty cycle; and the propellant distribution lines, isolation valves, and engine weights were optimized based on available pressure budget. Again, a 2 lbf/in^2 pressure differential was maintained across the engine injector and valve. To provide minimum weights main engine tank design pressures of 20 lbf/in^2a were used and the ratio of resupply propellant flow rate to main tank outflow rate was maintained at unity. The rationale for these design conditions are discussed in the body of this report. APS propellant tankage requirements were determined based on available residuals and mission total impulse requirements.

For the 10 and 50 ft/sec velocity cases, component and total weight summaries are presented for each subsystem concept in Figures F-17 through F-22 (orbiter A) and in Figures F-23 through F-28 (orbiter B). For reference purposes these figures also include weight breakdowns for the alternate flow control concepts. The all maneuver weight summaries are tabulated in Figure F-29. For the passive conditioning case, propellant, tank and engine weights were found to be nearly independent of flow control concepts. In contrast, propellant weights were reduced for active conditioning by utilizing constant density control since the lower conditioning requirements decreased the active heat exchanger/gas generator propellant usage. The constant density control concept also resulted in lower heat exchanger weights, again because of reduced conditioning energy requirements. Furthermore, distribution line weights are reduced because of the lower design operating temperatures of this control concept.

For the selected constant density flow control approach, the subsystem weights have again been summarized in Figure F-30 for orbiter A and Figure F-31 for orbiter B. Minimum subsystem weight was achieved for the 10 and 50 ft/sec maneuvers utilizing liquid propellant storage and passive conditioning. Minimum low pressure APS weights for the 50 ft/sec velocity allocation are 11,632 and 14,543 lb for orbiters A and B respectively.

. LIQUID STORAGE
. ≤10 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	799	792	792	802	952	943	943	845
OXYGEN (O_2)	2398	2376	2376	2406	2496	2472	2472	2438
HYDROGEN TANK								
PRESSURIZATION SYSTEM	122	122	122	122	126	126	126	123
TANK AND INSULATION	207	206	206	208	234	233	233	216
SCREEN	38	38	38	38	43	43	43	39
OXYGEN TANK								
PRESSURIZATION SYSTEM	16	16	16	17	17	16	16	16
TANK AND INSULATION	104	104	104	105	107	106	106	106
SCREEN	12	12	12	13	13	13	13	13
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	49	49	49	22
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	171	171	171	0	240	240	240	0
HEAT EXCHANGER (O_2)	331	331	331	280	250	250	250	232
DISTRIBUTION SYSTEM								
LINES	522	522	522	406	522	522	522	406
CONTROLS (VALVES & REGS)	358	722	620	662	291	755	683	686
ENGINES	1646	1646	1646	1613	1646	1646	1646	1613
TOTAL	6723	7058	6956	6672	7086	7414	7342	6755

FIGURE F-17

ORBITER A SUBSYSTEM WEIGHT SUMMARY

- SUPERCRITICAL STORAGE, NON-REFILLABLE
- ≤ 10 FPS

PARAMETER \ CONCEPT	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	881	873	873	915	988	978	978	915
OXYGEN (O_2)	2387	2364	2364	2486	2452	2429	2429	2486
HYDROGEN TANK								
PRESSURIZATION SYSTEM	195	182	182	167	208	208	208	182
TANK AND INSULATION SCREEN	439	435	435	435	490	485	485	455
OXYGEN TANK								
PRESSURIZATION SYSTEM	196	184	184	167	173	173	173	172
TANK AND INSULATION SCREEN	331	328	328	328	339	336	336	343
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	50	50	50	25
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	37	37	37	0	243	243	243	
HEAT EXCHANGER (O_2)	23	23	23	20	243	243	243	223
DISTRIBUTION SYSTEM								
LINES	521	521	521	405	521	521	521	405
CONTROLS (VALUES & REGS)	608	972	900	885	641	1005	933	909
ENGINES	1646	1646	1646	1613	1646	1646	1646	1613
TOTAL	7264	7565	7493	7421	7994	8317	8245	7728

FIGURE F-18

ORBITER A SUBSYSTEM WEIGHT SUMMARY

- SUPERCRITICAL STORAGE - REFILLABLE
- ≤10 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	887	879	879	921	994	984	984	921
OXYGEN (O_2)	2395	2372	2372	2494	2460	2437	2437	2494
HYDROGEN TANK								
PRESSURIZATION SYSTEM	195	182	182	167	207	208	208	182
TANK AND INSULATION SCREEN	237	237	237	237	237	237	237	237
OXYGEN TANK								
PRESSURIZATION SYSTEM	196	184	184	167	173	173	173	172
TANK AND INSULATION SCREEN	148	148	148	148	148	148	148	148
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	50	50	50	25
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	37	37	37	–	243	243	243	
HEAT EXCHANGER (O_2)	23	23	23	20	243	243	243	223
DISTRIBUTION SYSTEM								
LINES	522	522	522	405	522	522	522	405
CONTROLS (VALVES & REGS)	608	972	900	855	641	1005	933	909
ENGINES	1646	1646	1646	1613	1646	1646	1646	1613
TOTAL	6894	7202	7130	7027	7564	7896	7824	7329

FIGURE F-19

ORBITER A SUBSYSTEM WEIGHT SUMMARY

. LIQUID STORAGE
. ≤50 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	1839	1829	1829	1846	2294	2281	2281	2019
OXYGEN (O_2)	5520	5486	5486	5539	5746	5711	5711	5627
HYDROGEN TANK								
PRESSURIZATION SYSTEM	143	143	143	143	150	150	150	146
TANK AND INSULATION	388	386	386	389	470	468	468	421
SCREEN	68	68	68	69	80	80	80	74
OXYGEN TANK								
PRESSURIZATION SYSTEM	36	36	36	36	38	38	38	37
TANK AND INSULATION	165	165	165	165	169	168	168	167
SCREEN	22	22	22	22	23	23	23	22
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	53	53	53	45
GAS GENERATOR (O_2)	0	0	0	0				
HEAT EXCHANGER (H_2)	734	734	734	162	276	276	276	128
HEAT EXCHANGER (O_2)	606	610	610	491	280	282	282	255
DISTRIBUTION SYSTEM								
LINES	522	522	522	405	522	522	522	405
CONTROLS (VALVES & REGS)	358	722	620	662	391	755	683	686
ENGINES	1734	1734	1734	1703	1734	1734	1734	1703
TOTAL	12135	12457	12355	11632	12226	12541	12469	11735

FIGURE F-20

ORBITER A SUBSYSTEM WEIGHT SUMMARY

. SUPERCRITICAL STORAGE, NON- REFILLABLE
. ≤50 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	2012	2000	2000	2019	2356	2342	2342	2044
OXYGEN (O_2)	5465	5431	5431	5486	5630	5595	5595	5498
HYDROGEN TANK								
PRESSURIZATION SYSTEM	109	109	109	123	221	220	220	186
TANK AND INSULATION SCREEN	968	962	962	971	1127	1121	1121	983
OXYGEN TANK								
PRESSURIZATION SYSTEM	105	105	105	117	176	176	176	169
TANK AND INSULATION SCREEN	707	703	703	710	727	723	723	711
THERMAL CONDITIONING								
GAS GENERATOR (H_2)					54	54	54	45
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	226	226	226	48	284	284	284	130
HEAT EXCHANGER (O_2)	162	162	162	97	147	247	247	222
DISTRIBUTION SYSTEM								
LINES	522	522	522	405	522	522	522	405
CONTROLS (**VALVES** & REGS)	608	972	900	885	641	1005	933	909
ENGINES	1734	1734	1734	1703	1734	1734	1734	1703
TOTAL	12618	12926	12854	12564	13719	14023	13951	13005

FIGURE F-21

ORBITER A SUBSYSTEM WEIGHT SUMMARY

. SUPERCRITICAL STORAGE REFILLABLE
. ≤50 FPS

PARAMETER \ CONCEPT	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	2018	2006	2006	2025	2362	2348	2348	2050
OXYGEN (O_2)	5473	5439	5439	5494	5638	5603	5603	5506
HYDROGEN TANK								
PRESSURIZATION SYSTEM	109	109	109	123	221	220	220	186
TANK AND INSULATION SCREEN	523	523	523	523	523	523	523	523
OXYGEN TANK								
PRESSURIZATION SYSTEM	105	105	105	117	176	176	176	169
TANK AND INSULATION SCREEN	342	342	342	342	342	342	342	342
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	54	54	54	45
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	226	226	226	48	284	284	284	130
HEAT EXCHANGER (O_2)	162	162	162	97	247	247	247	222
DISTRIBUTION SYSTEM								
LINES	522	522	522	405	522	522	522	405
CONTROLS (VALVES & REGS)	608	972	900	885	641	1005	933	909
ENGINES	1734	1734	1734	1703	1734	1734	1734	1703
TOTAL	11822	12140	12068	11762	12744	13058	12986	12190

FIGURE F-22

ORBITER A SUBSYSTEM WEIGHT SUMMARY

. LIQUID STORAGE
. ≤ 10 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	1030	1028	1028	1037	1311	1324	1324	1149
OXYGEN (O_2)	3091	3085	3085	3112	3289	3294	3294	3199
HYDROGEN TANK								
PRESSURIZATION SYSTEM	127	127	127	128	133	133	133	130
TANK AND INSULATION	247	247	247	249	293	295	295	267
SCREEN	45	45	45	47	54	54	54	50
OXYGEN TANK								
PRESSURIZATION SYSTEM	23	23	23	23	24	24	24	24
TANK AND INSULATION	120	120	120	120	124	124	124	122
SCREEN	15	15	15	15	16	16	16	15
THERMAL CONDITIONING								
GAS GENERATOR (H_2)								
GAS GENERATOR (O_2)	0	0	0	0	63	65	65	47
HEAT EXCHANGER (H_2)	475	507	507	173	523	562	562	230
HEAT EXCHANGER (O_2)	637	637	637	413	303	305	305	242
DISTRIBUTION SYSTEM								
LINES	580	580	580	450	580	580	580	450
CONTROLS (VALVES & REGS)	458	952	794	875	508	1002	880	910
ENGINES	2130	2130	2130	2092	2130	2131	2131	2092
TOTAL	8978	9496	9338	8734	9351	9909	9787	8927

FIGURE F-23

ORBITER B SUBSYSTEM WEIGHT SUMMARY

. SUPERCRITICAL STORAGE (NON REFILLABLE)
. ≤10 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING MASS ADD	PASSIVE CONDITIONING SUPPLY REG	PASSIVE CONDITIONING THRUSTER REG	PASSIVE CONDITIONING P/T	ACTIVE CONDITIONING MASS ADD	ACTIVE CONDITIONING SUPPLY REG	ACTIVE CONDITIONING THRUSTER REG	ACTIVE CONDITIONING P/T
PROPELLANT								
HYDROGEN (H_2)	1137	1135	1135	1144	1362	1376	1376	1144
OXYGEN (O_2)	3083	3077	3077	3110	3235	3240	3240	3110
HYDROGEN TANK								
PRESSURIZATION SYSTEM	487	522	522	528	579	588	588	528
TANK AND INSULATION SCREEN	560	599	599	605	665	672	672	605
OXYGEN TANK								
PRESSURIZATION SYSTEM	429	429	429	435	449	451	451	435
TANK AND INSULATION SCREEN	417	416	416	420	436	436	436	420
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	65	66	66	48
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	109	118	118	51	503	517	517	220
HEAT EXCHANGER (O_2)	28	28	28	18	373	374	374	276
DISTRIBUTION SYSTEM								
LINES	580	580	580	450	580	580	580	450
CONTROLS (VALVES & REGS)	863	1357	1199	1217	913	1407	1285	1252
ENGINES	2131	2131	2131	2092	2131	2131	2131	2092
TOTAL	9824	10392	10234	10070	11291	11838	11716	10580

FIGURE F-24

ORBITER B SUBSYSTEM WEIGHT SUMMARY

. SUPERCRITICAL STORAGE (REFILLABLE)
. ≤10 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	1143	1145	1145	1152	1368	1384	1384	1152
OXYGEN (O_2)	3091	3085	3085	3118	3243	3248	3248	3118
HYDROGEN TANK								
PRESSURIZATION SYSTEM	487	522	522	528	579	588	588	528
TANK AND INSULATION SCREEN	281	281	281	281	281	281	281	281
OXYGEN TANK								
PRESSURIZATION SYSTEM	429	429	429	435	449	451	451	435
TANK AND INSULATION SCREEN	175	175	175	175	175	175	175	175
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	65	66	66	48
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	109	118	118	51	503	517	517	220
HEAT EXCHANGER (O_2)	28	28	28	18	373	374	374	276
DISTRIBUTION SYSTEM								
LINES	580	580	580	450	580	580	580	450
CONTROLS (VALVES & REGS)	863	1357	1199	1217	913	1407	1285	1252
ENGINES	2131	2131	2131	2092	2131	2131	2131	2092
TOTAL	9317	9851	9693	9447	10660	11202	11080	10027

FIGURE F-25

ORBITER B SUBSYSTEM WEIGHT SUMMARY

- LIQUID STORAGE
- ≤ 50 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	2246	2248	2248	2232	2950	2955	2955	2534
OXYGEN (O_2)	6739	6743	6743	6696	7095	7100	7100	6861
HYDROGEN TANK								
PRESSURIZATION SYSTEM	149	149	149	149	160	160	160	154
TANK AND INSULATION	438	439	439	436	556	556	556	487
SCREEN	78	78	78	78	96	96	96	86
OXYGEN TANK								
PRESSURIZATION SYSTEM	50	50	50	49	52	52	52	50
TANK AND INSULATION	184	185	185	184	190	190	190	187
SCREEN	25	25	25	25	26	26	26	25
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	70	70	70	53
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	1298	1312	1312	463	596	604	604	269
HEAT EXCHANGER (O_2)	829	818	818	699	311	299	299	277
DISTRIBUTION SYSTEM								
LINES	580	580	580	451	580	580	580	451
CONTROLS (VALVES & REGS)	458	952	794	875	508	1002	880	910
ENGINES	2247	2247	2247	2206	2247	2247	2247	2206
TOTAL	15321	15826	15668	14543	15437	15937	15815	14550

FIGURE F-26

ORBITER 3 SUBSYSTEM WEIGHT SUMMARY

- SUPERCRITICAL STORAGE (NON REFILLABLE)
- $\leq$ 50 FPS

CONCEPT / PARAMETER	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	2459	2461	2461	2440	3055	3062	3062	2578
OXYGEN (O_2)	6676	6679	6679	6621	6965	6970	6970	6694
HYDROGEN TANK								
PRESSURIZATION SYSTEM	495	497	497	500	609	611	611	519
TANK AND INSULATION SCREEN	1175	1175	1175	1179	1448	1452	1452	1243
OXYGEN TANK								
PRESSURIZATION SYSTEM	445	445	445	445	447	447	447	446
TANK AND INSULATION SCREEN	853	853	853	857	888	889	889	866
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	72	72	72	54
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	382	387	387	135	572	575	575	260
HEAT EXCHANGER (O_2)	155	155	155	116	371	371	371	333
DISTRIBUTION SYSTEM								
LINES	580	580	580	450	580	580	580	450
CONTROLS (VALVES & REGS)	863	1357	1199	1217	913	1407	1285	1252
ENGINES	2247	2247	2247	2206	2247	2247	2247	2206
TOTAL	16330	16836	16678	16166	18167	18683	18561	16901

ORBITER B SUBSYSTEM WEIGHT SUMMARY

FIGURE F-27

. SUPERCRITICAL STORAGE (REFILLABLE)
. ≤ 50 FPS

PARAMETER \ CONCEPT	PASSIVE CONDITIONING				ACTIVE CONDITIONING			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	2465	2476	2467	2440	3061	3068	3068	2578
OXYGEN (O_2)	6684	6687	6687	6621	6973	6978	6978	6694
HYDROGEN TANK								
PRESSURIZATION SYSTEM	495	497	497	498	609	611	611	519
TANK AND INSULATION SCREEN	620	620	620	620	620	620	620	620
OXYGEN TANK								
PRESSURIZATION SYSTEM	445	445	445	445	447	447	447	446
TANK AND INSULATION SCREEN	406	406	406	406	406	406	406	406
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	0	0	0	0	72	72	72	54
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	382	387	387	135	572	575	575	260
HEAT EXCHANGER (O_2)	155	155	155	116	371	371	371	333
DISTRIBUTION SYSTEM								
LINES	580	580	580	450	580	580	580	450
CONTROLS (VALVES & REGS)	863	1357	1199	1217	913	1407	1285	1252
ENGINES	2247	2247	2247	2206	2247	2247	2247	2206
TOTAL	15342	15848	15690	15154	16871	17382	17260	15818

FIGURE F-28

ORBITER B SUBSYSTEM WEIGHT SUMMARY

. LIQUID STORAGE/ACTIVE CONDITIONING
. ALL MANEUVERS

PARAMETER \ CONCEPT	ORBITER A				ORBITER B			
	MASS ADD	SUPPLY REG	THRUSTER REG	P/T	MASS ADD	SUPPLY REG	THRUSTER REG	P/T
PROPELLANT								
HYDROGEN (H_2)	11572	11511	11511	10643	13901	13867	13867	12526
OXYGEN (O_2)	28253	28126	28126	28029	33037	32882	32882	32538
HYDROGEN TANK								
PRESSURIZATION SYSTEM	233	233	233	252	239	239	239	235
TANK AND INSULATION	1997	1987	1987	1850	2216	2212	2212	2016
SCREEN	232	232	232	220	263	263	263	245
OXYGEN TANK								
PRESSURIZATION SYSTEM	181	181	181	180	238	236	236	234
TANK AND INSULATION	423	423	423	422	470	467	467	464
SCREEN	66	65	65	65	73	72	72	72
THERMAL CONDITIONING								
GAS GENERATOR (H_2)	55	55	55	49	72	73	73	59
GAS GENERATOR (O_2)								
HEAT EXCHANGER (H_2)	289	287	287	178	597	606	606	340
HEAT EXCHANGER (O_2)	294	294	294	278	335	345	345	326
DISTRIBUTION SYSTEM								
LINES	522	522	522	405	580	580	580	451
CONTROLS (VALVES & REGS)	391	755	683	686	508	1002	880	910
ENGINES	1734	1734	1734	1703	2247	2247	2247	2206
TOTAL	46242	46405	46333	44960	54776	55091	54969	52622

FIGURE F-29

ORBITER A & B SUBSYSTEM WEIGHT SUMMARY

PARAMETER \ CONCEPT	≤10 ft/sec				≤50 ft/sec				All Maneuvers
	SUPERCRITICAL		LIQUID		SUPERCRITICAL		LIQUID		LIQUID
	PASSIVE	ACTIVE	PASSIVE	ACTIVE	PASSIVE	ACTIVE	PASSIVE	ACTIVE	ACTIVE
PROPELLANT									
HYDROGEN (H_2)	915	915	802	845	2025	2050	1845	2019	10643
OXYGEN (O_2)	2486	2486	2406	2438	5494	5506	5539	5627	28029
HYDROGEN TANK									
PRESSURIZATION SYSTEM	167	182	122	123	123	186	143	146	252
TANK AND INSULATION	435	455	208	216	523	523	389	421	1850
SCREEN			38	39			69	74	220
OXYGEN TANK									
PRESSURIZATION SYSTEM	167	172	17	16	117	169	36	37	180
TANK AND INSULATION	328	343	105	106	342	342	165	167	422
SCREEN			13	13			22	22	65
THERMAL CONDITIONING									
GAS GENERATOR (H_2)	0	25	0	22	0	45	0	45	49
GAS GENERATOR (O_2)							0		
HEAT EXCHANGER (H_2)	0		0	0	48	130	162	128	178
HEAT EXCHANGER (O_2)	20	223	280	232	97	222	491	255	278
DISTRIBUTION SYSTEM									
LINES	405	405	406	406	405	405	405	405	405
CONTROLS (VALVES & REGS)	885	909	662	686	885	909	662	686	686
ENGINES	1613	1613	1613	1613	1703	1703	1703	1703	1703
TOTAL	7421	7728	6672	6755	11762	12190	11632	11735	44960

ORBITER A WEIGHT SUMMARY
°P/T CONTROL

FIGURE F-30

PARAMETER \ CONCEPT	≤ 10 FT/SEC				≤ 50 FT/SEC				ALL MANEUV.
	SUPERCRITICAL		LIQUID		SUPERCRITICAL		LIQUID		LIQUID
	PASS.	ACT.	PASS.	ACT.	PASS.	ACT.	PASS.	ACT.	ACT.
PROPELLANT									
HYDROGEN (H_2)	1144	1144	1037	1149	2440	2578	2232	2534	12526
OXYGEN (O_2)	3110	3110	3112	3199	6621	6694	6696	6871	32538
HYDROGEN TANK									
PRESSURIZATION SYSTEM	528	528	128	130	498	519	149	154	235
TANK AND INSULATION	605	605	249	267	620	620	436	487	2016
SCREEN	0	0	47	50	0	0	78	86	245
OXYGEN TANK									
PRESSURIZATION SYSTEM	435	435	23	24	445	446	49	50	234
TANK AND INSULATION	420	420	120	122	406	406	184	187	464
SCREEN	0	0	15	15	0	0	25	25	72
THERMAL CONDITIONING									
GAS GENERATORS	0	48	0	47	0	54	0	53	59
HEAT EXCHANGER (H_2)	51	220	173	230	135	260	463	269	340
HEAT EXCHANGER (H_2)	18	276	413	242	116	333	699	277	326
DISTRIBUTION SYSTEM									
LINES	450	450	450	450	450	450	451	451	451
CONTROLS (VALVES & REGS)	1217	1252	875	910	1217	1252	875	910	910
ENGINES	2092	2092	2092	2092	2206	2206	2206	2206	2206
TOTAL	10070	10580	8734	8927	15154	15818	14543	14550	52622

ORBITER B WEIGHT SUMMARY
°P/T CONTROL

FIGURE F-31

F-7. DESIGN AND WEIGHT SENSITIVITIES - ORBITER

The low pressure APS was investigated to determine subsystem design point and weight sensitivity to variations in design parameters. These variables included thrust, impulse, engine chamber pressure, mixture ratio, nozzle expansion ratio, maximum main engine tank pressures, and resupply propellant conditioning temperatures. As discussed for the booster, only linear sensitivities were evaluated. The resulting sensitivities are shown in Figures F-32 through F-40 for orbiter A and Figures F-41 through F-49 for orbiter B.

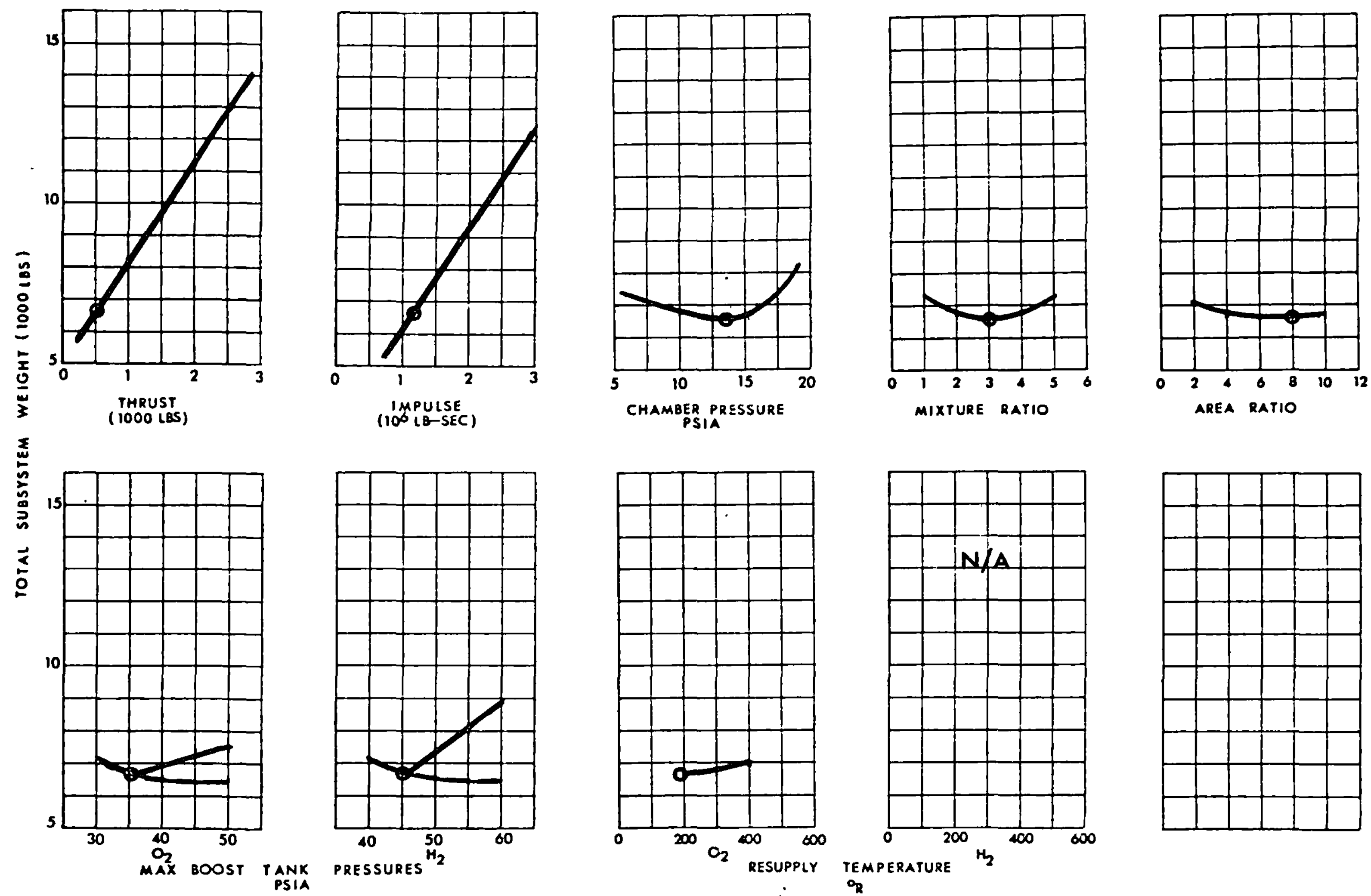
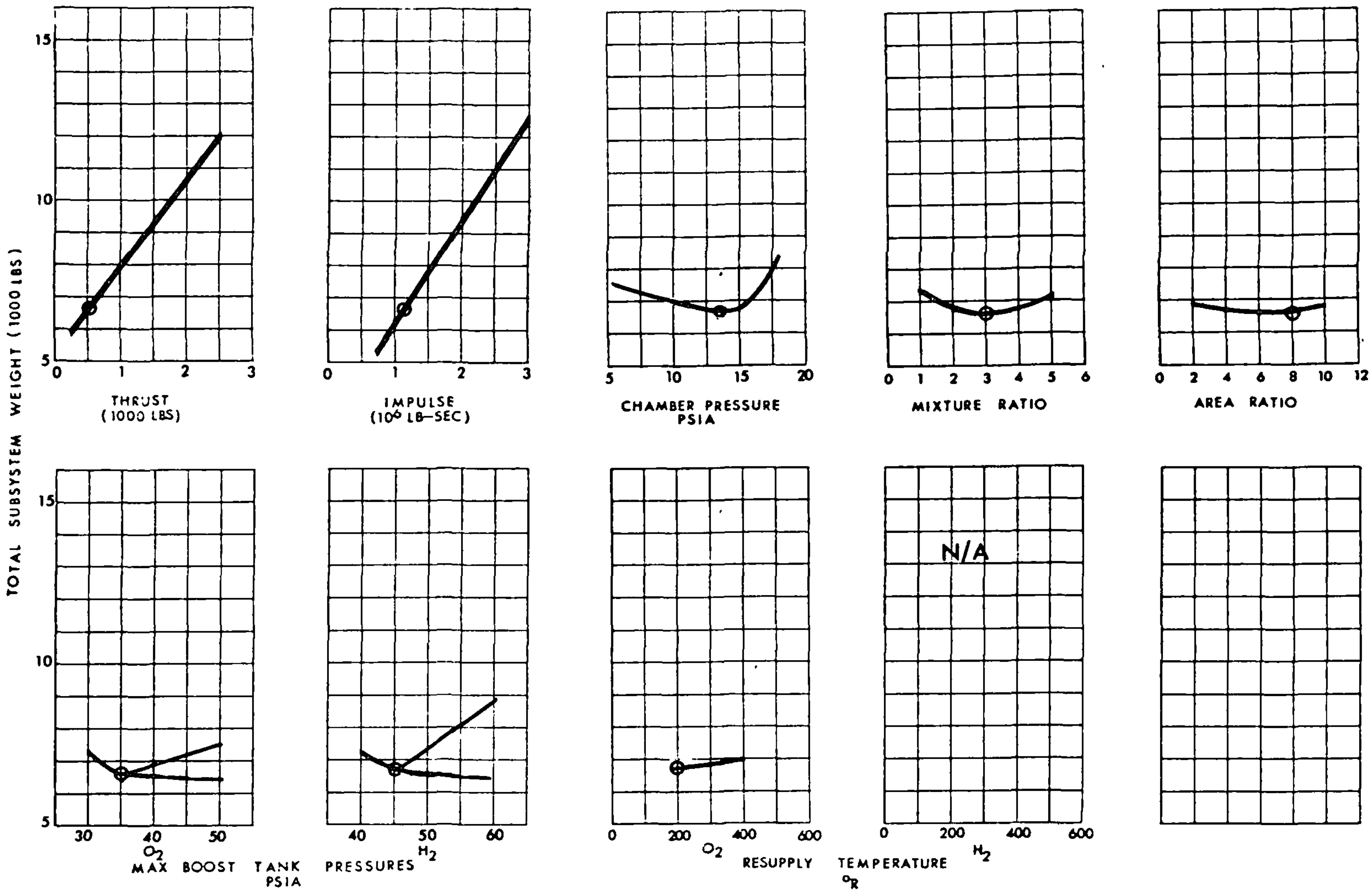


FIGURE F-32

LOW PRESSURE APS

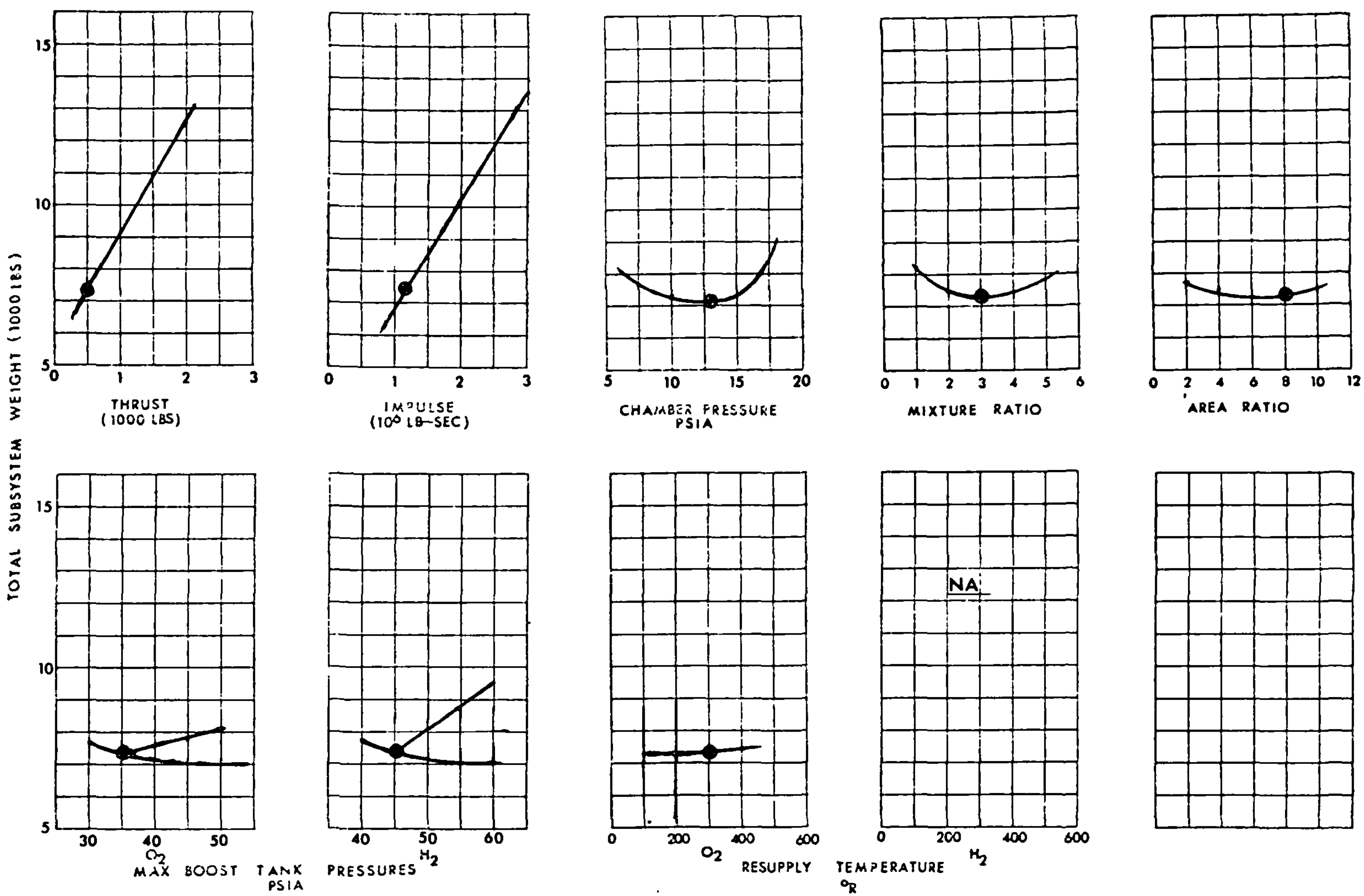
ORBITER A ($\leq$ 10 FPS)
LIQUID — PASSIVE HEX



LOW PRESSURE APS

ORBITER A (≤ 10 FPS)
LIQUID-ACTIVE HEX

FIGURE F-33

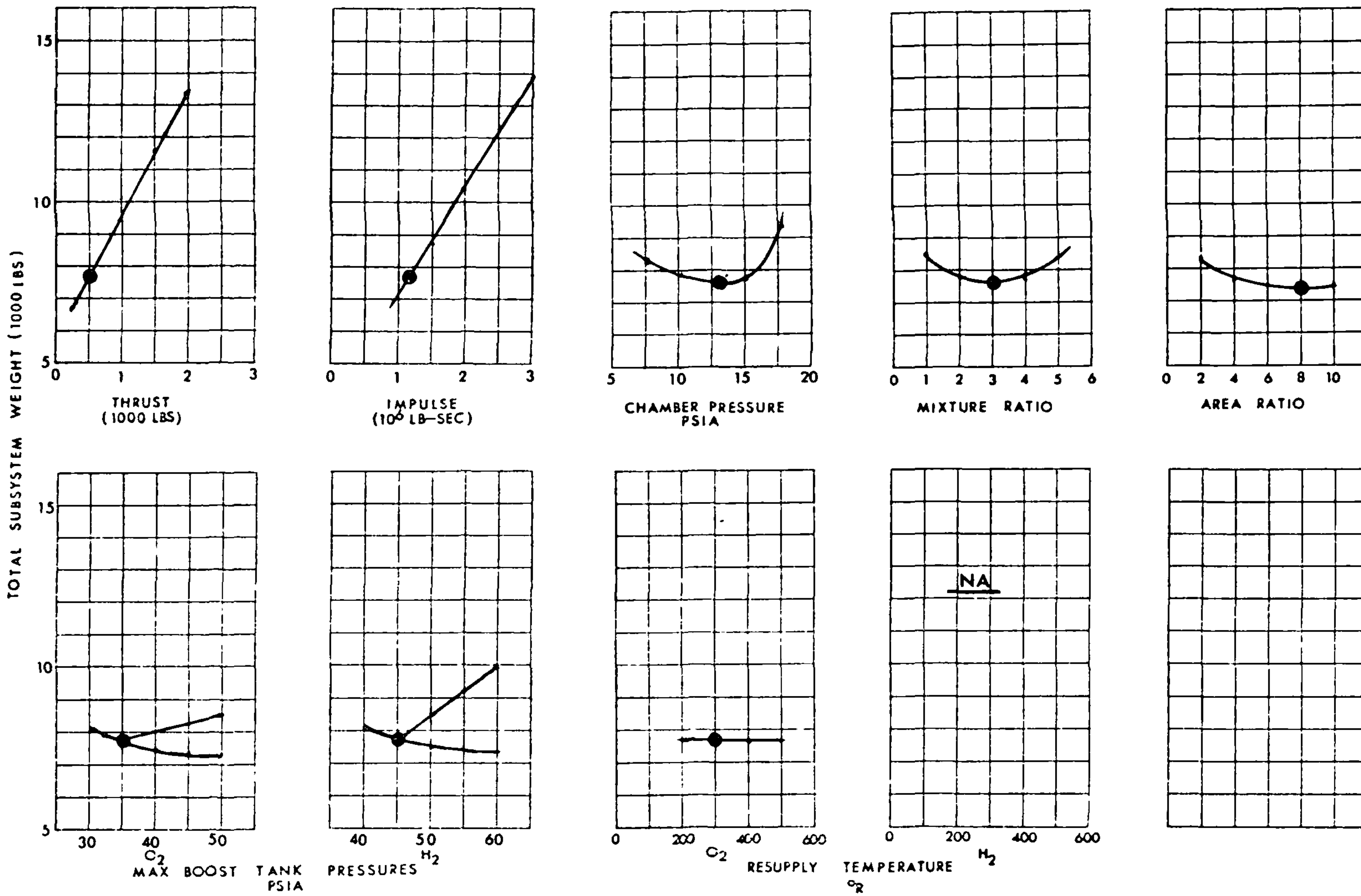


LOW PRESSURE APS

ORBITER A ($\leq$10FPS)

SUPERCRITICAL—PASSIVE

FIGURE F-34

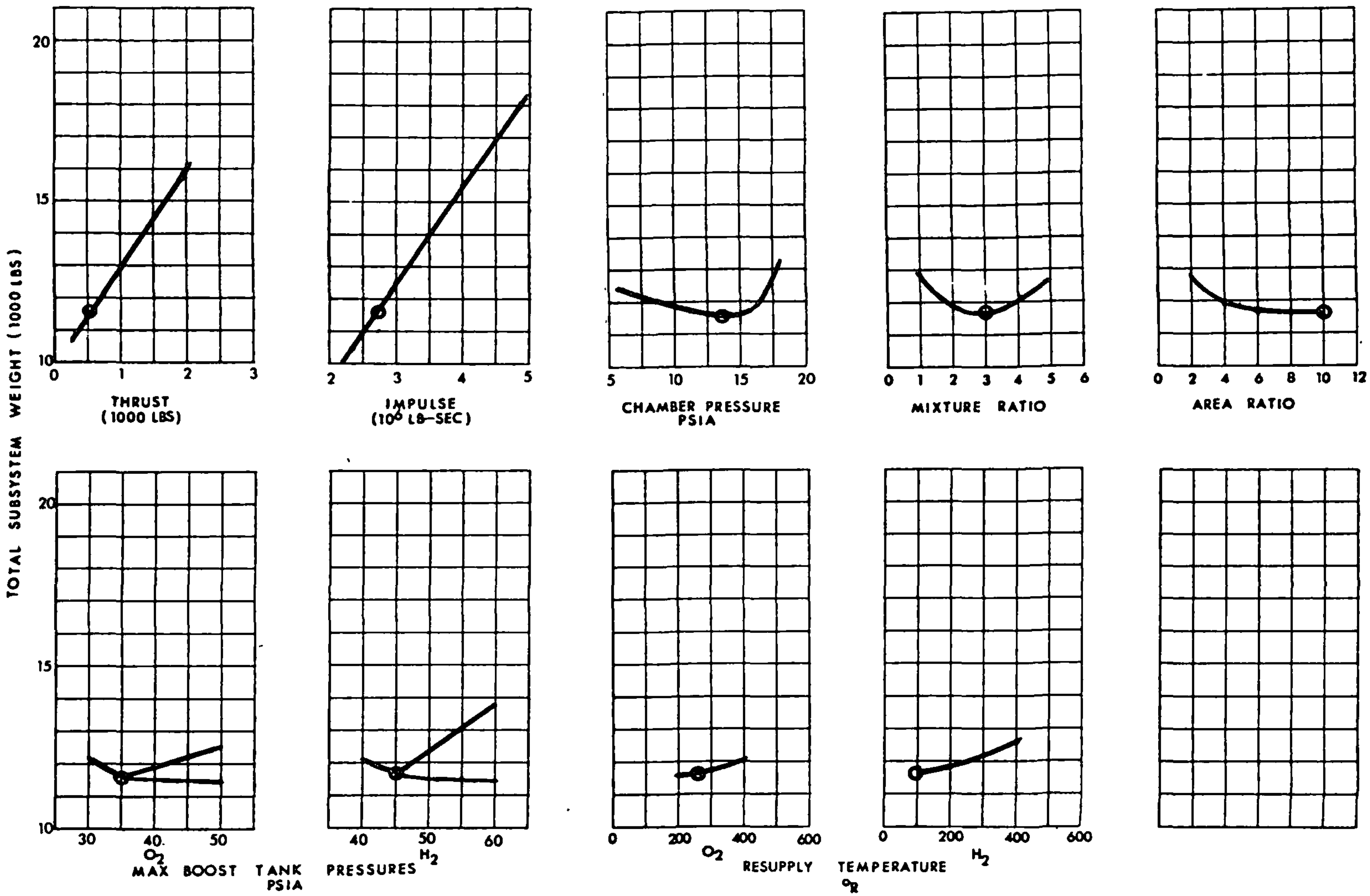


LOW PRESSURE APS

ORBITER A (≤ 10 FPS)

SUPERCRITICAL– ACTIVE

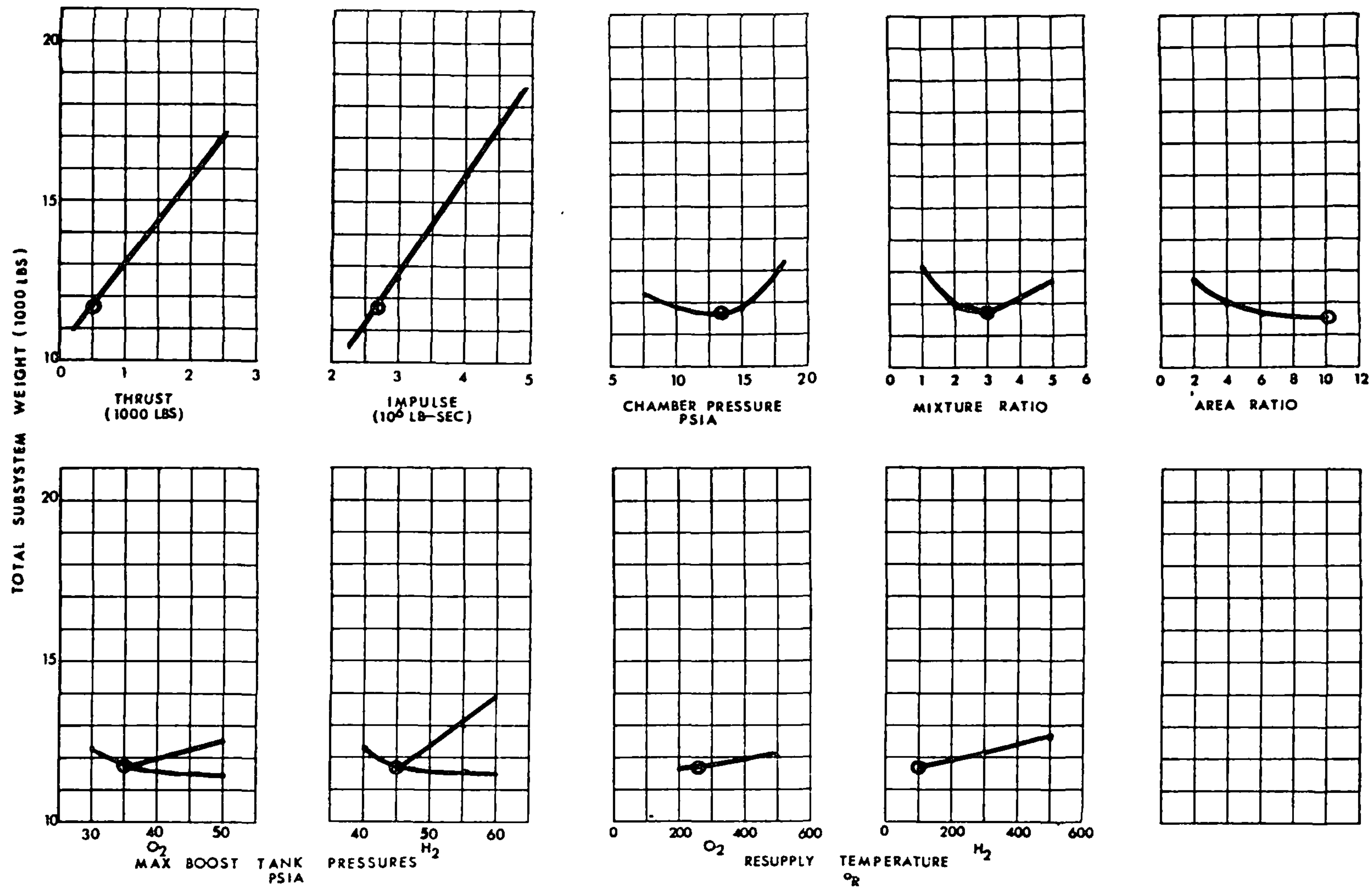
FIGURE F-35



LOW PRESSURE APS

ORBITER A ($\leq$50 FPS)
LIQUID – PASSIVE HEX

FIGURE F-36

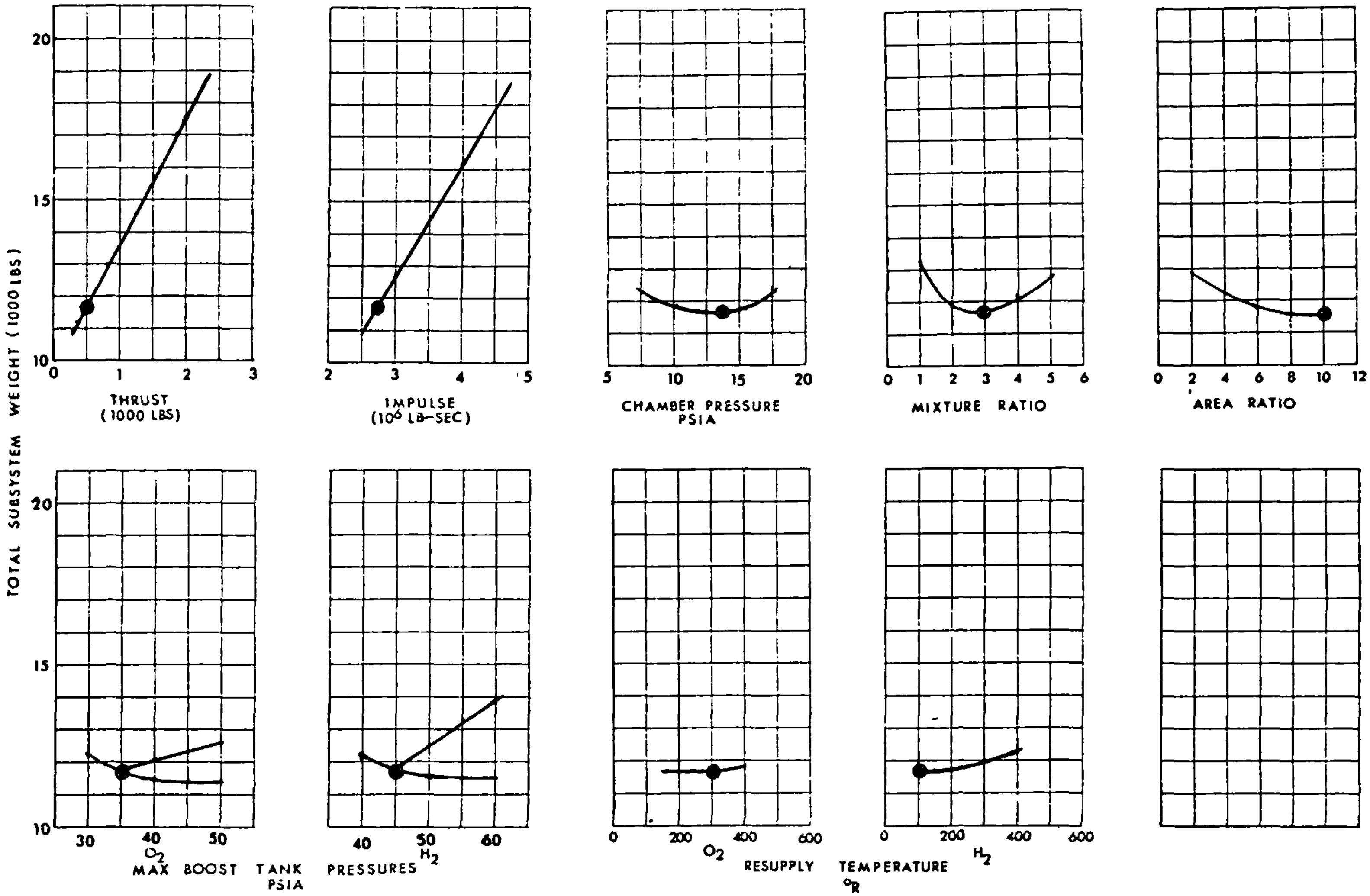


LOW PRESSURE APS

ORBITER A (≤ 50 FPS)

LIQUID– ACTIVE HEX

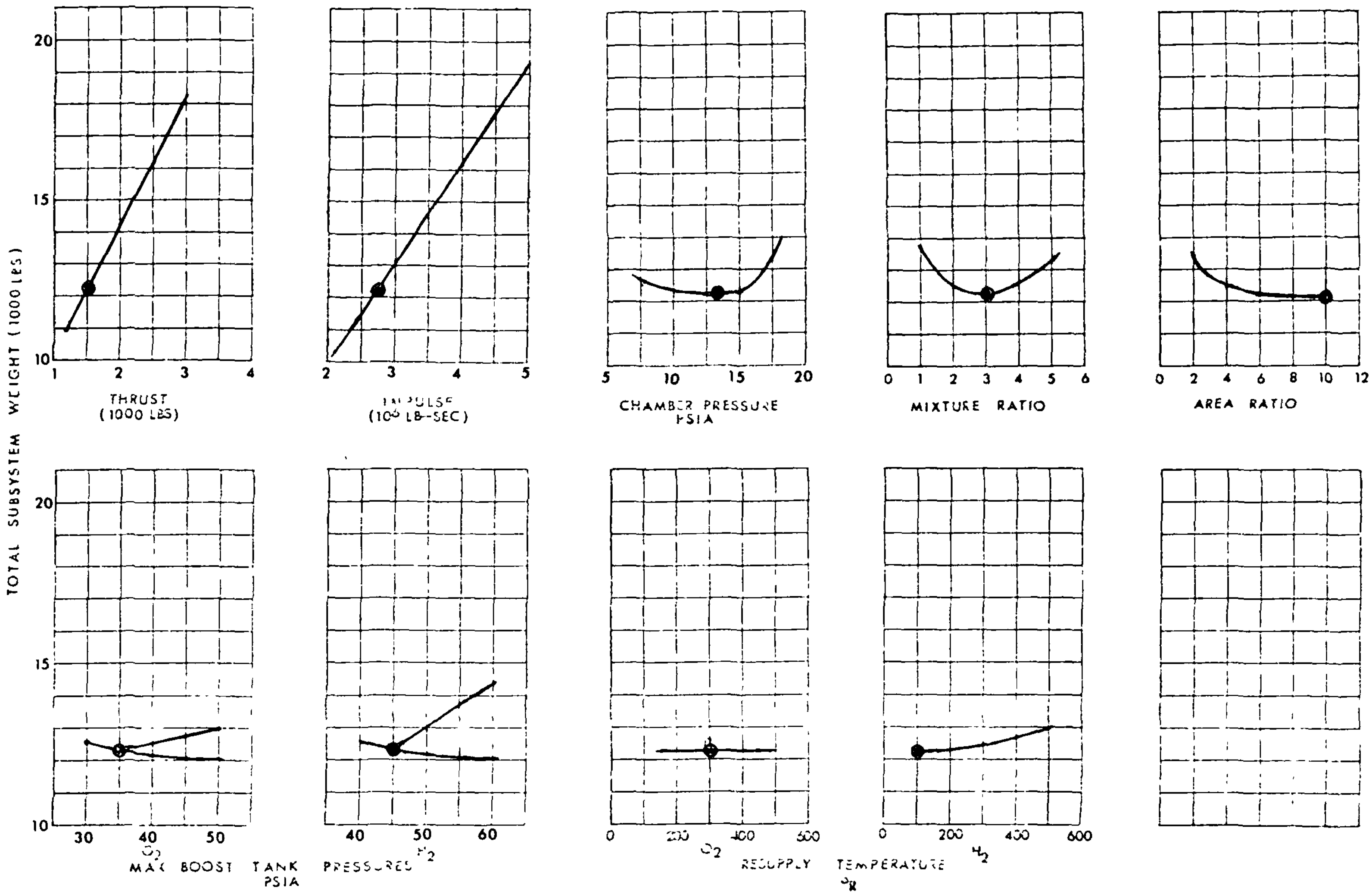
FIGURE F-37



LOW PRESSURE APS

ORBITER A ($\leq$50 FPS)
SUPERCRITICAL—PASSIVE

FIGURE F-38

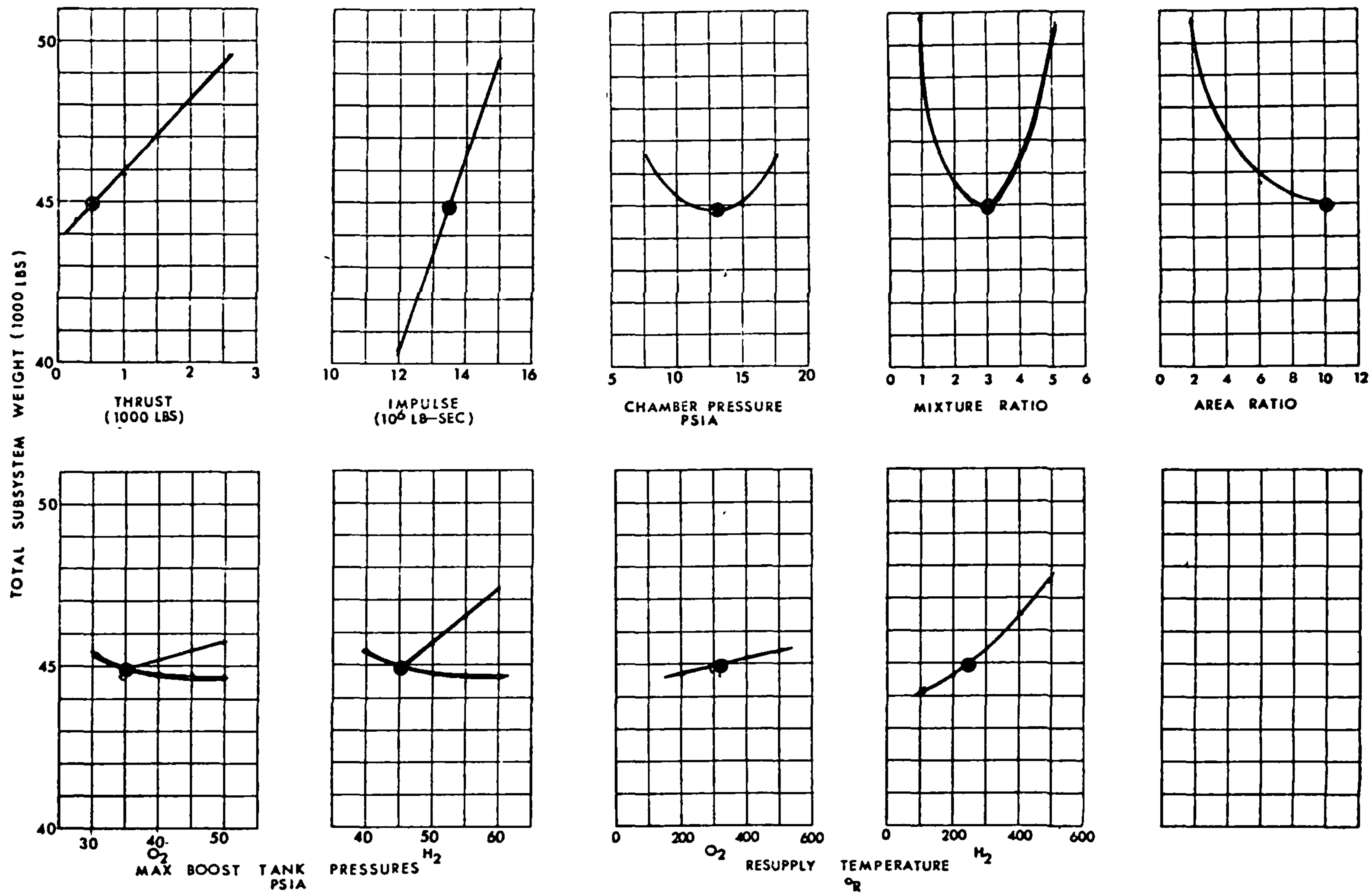


LOW PRESSURE APS

ORBITER A (≤50 FPS)

SUPERCRITICAL—ACTIVE

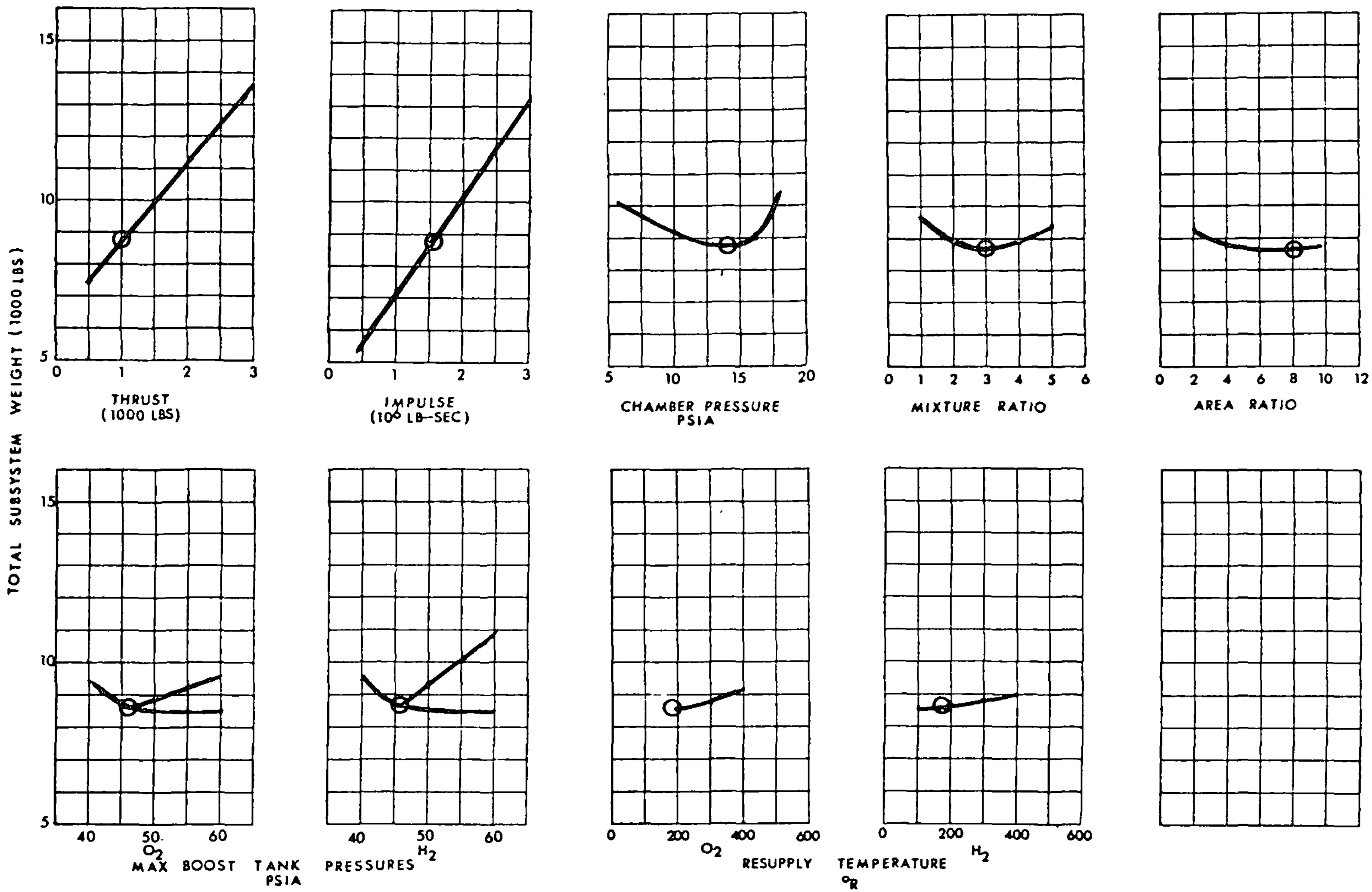
FIGURE F-39



LOW PRESSURE APS

ORBITER A (ALL MANEUVERS)
LIQUID–ACTIVE HEX

FIGURE F-40

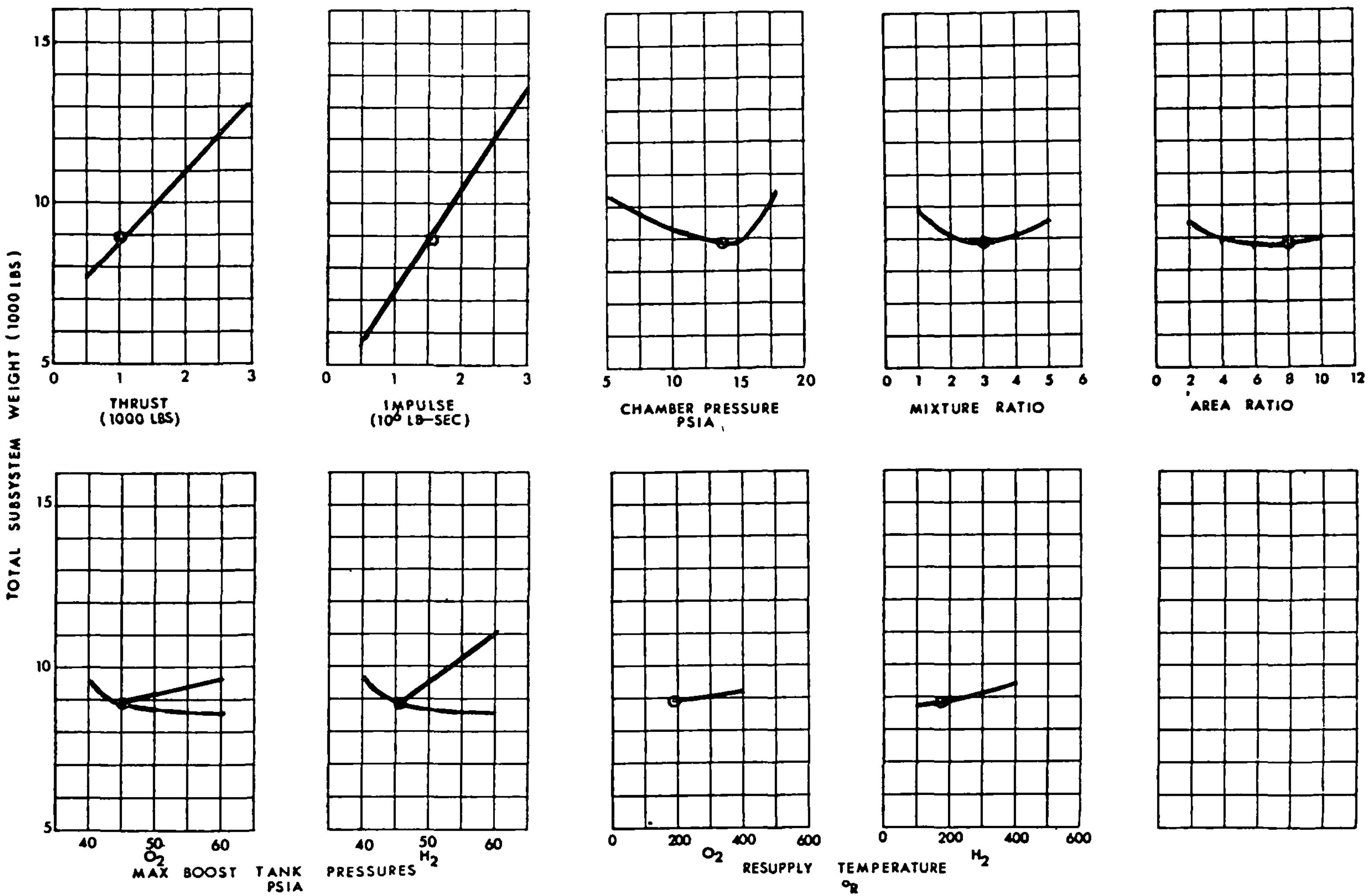


LOW PRESSURE APS

ORBITER B (≤10 FPS)

LIQUID – PASSIVE HEX

FIGURE F-41

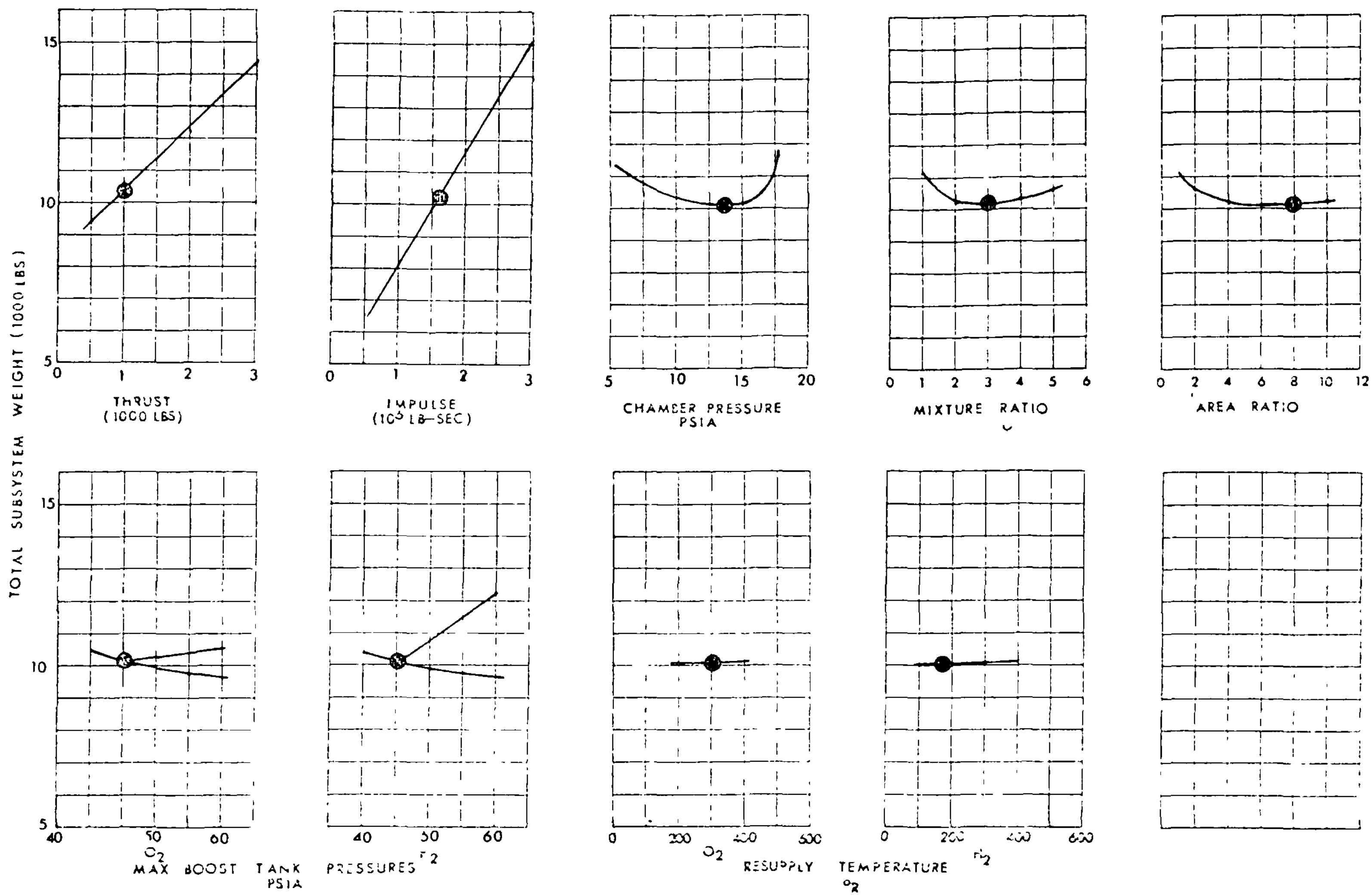


LOW PRESSURE APS

ORBITER B (≤ 10 FPS)

LIQUID – ACTIVE HEX

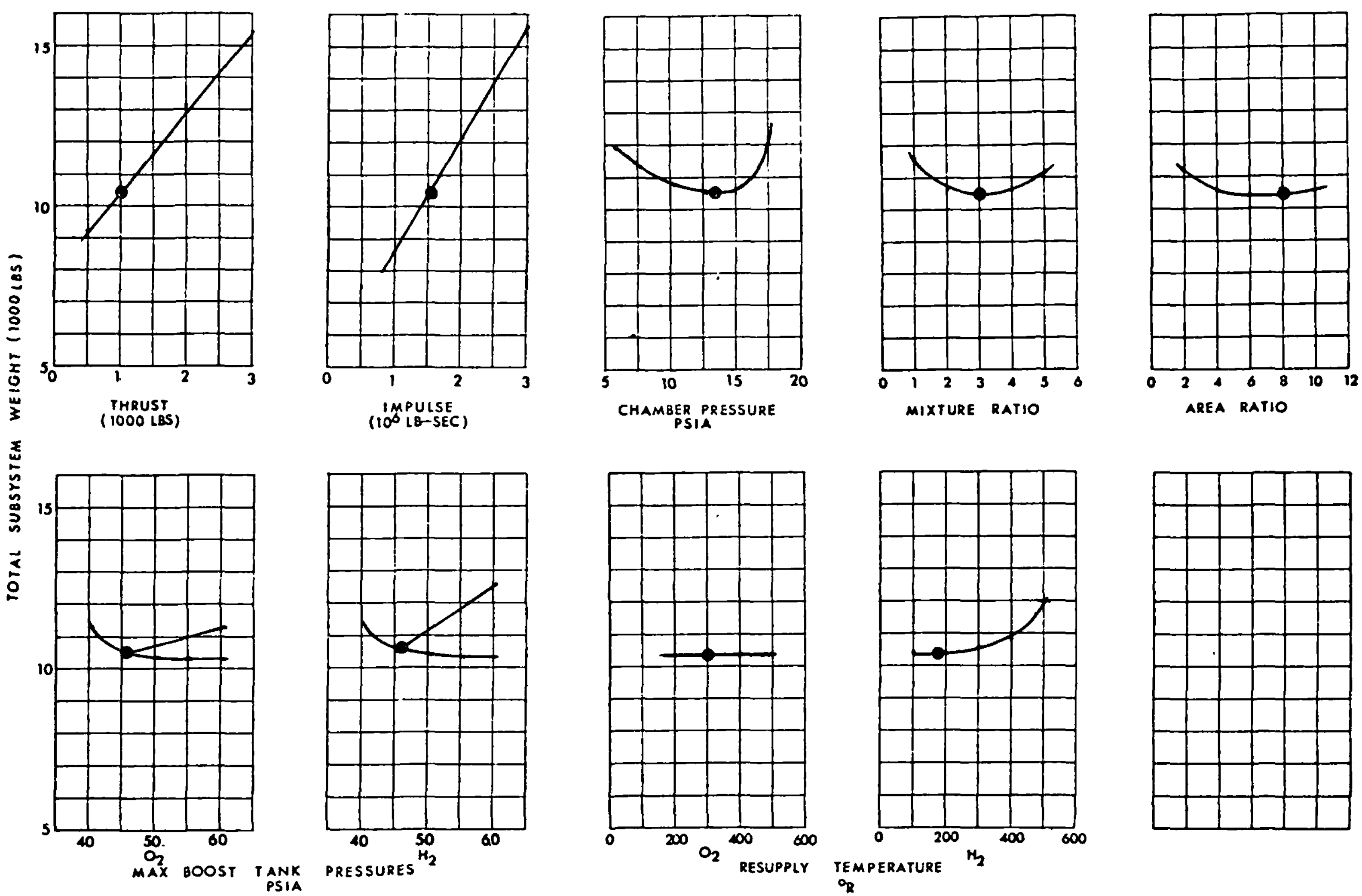
FIGURE F-42



LOW PRESSURE APS

ORBITER B (≤10 FPS)
SUPERCRITICAL—PASSIVE

FIGURE F-43

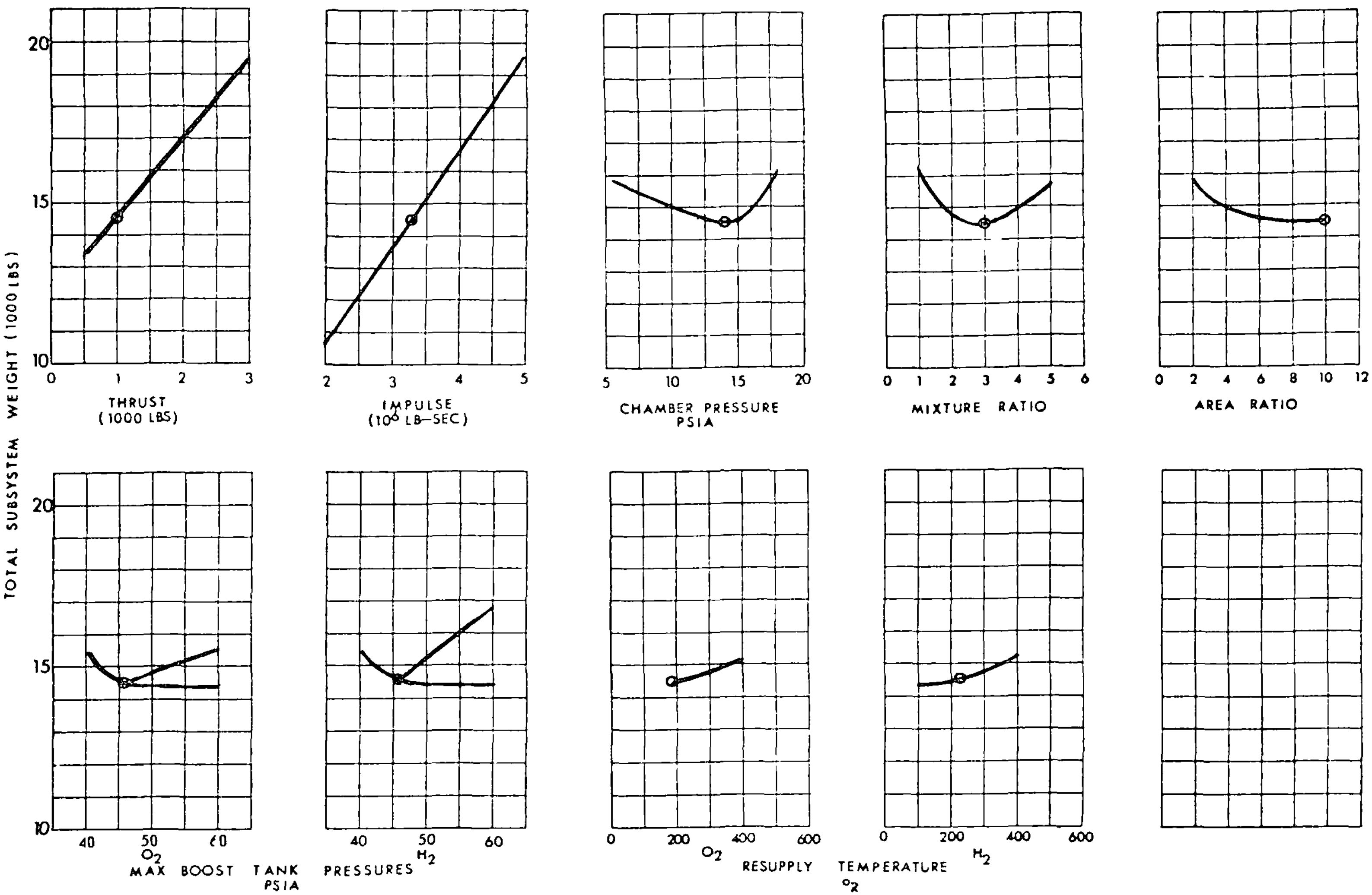


LOW PRESSURE APS

ORBITER B (≤10 FPS)

SUPERCRITAL-ACTIVE

FIGURE F-44

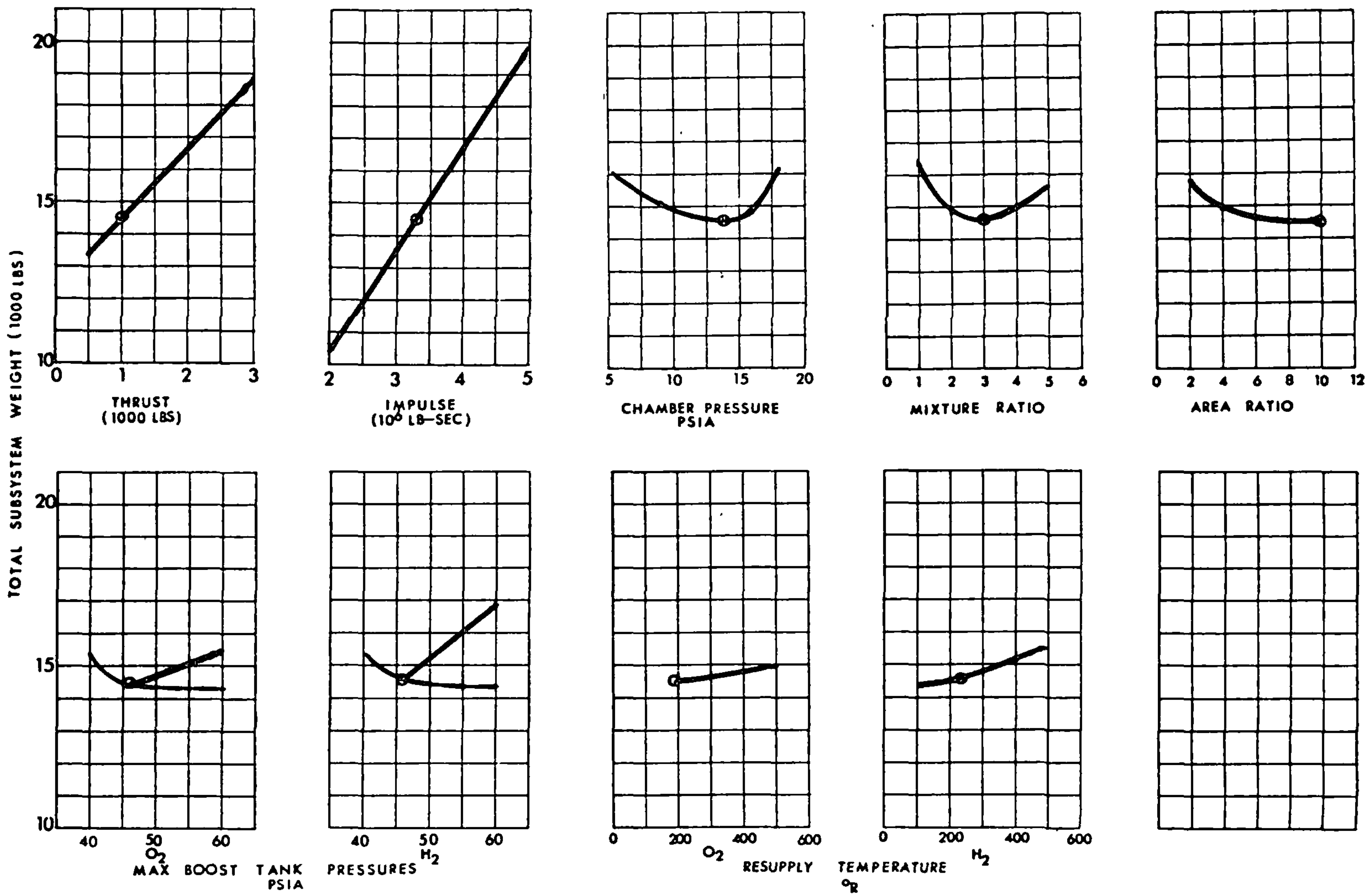


LOW PRESSURE APS

ORBITER B (≤50 FPS)

LIQUID—PASSIVE HEX

FIGURE F-45

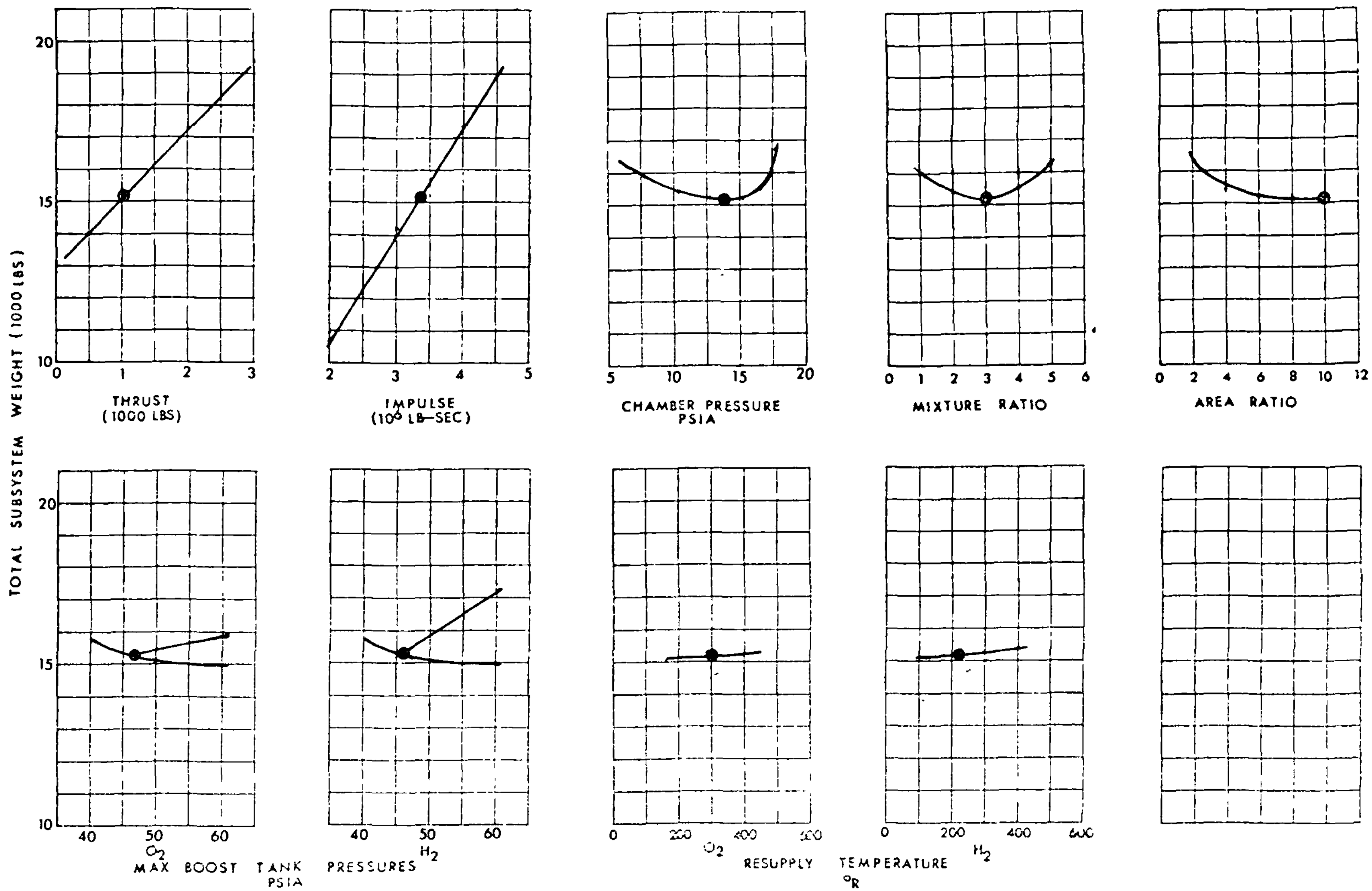


LOW PRESSURE APS

ORBITER B (≤50 FPS)

LIQUID–ACTIVE HEX

FIGURE F-46



LOW PRESSURE APS

ORBITER B (≤50 FPS)

SUPERCRITICAL – PASSIVE

FIGURE F-47

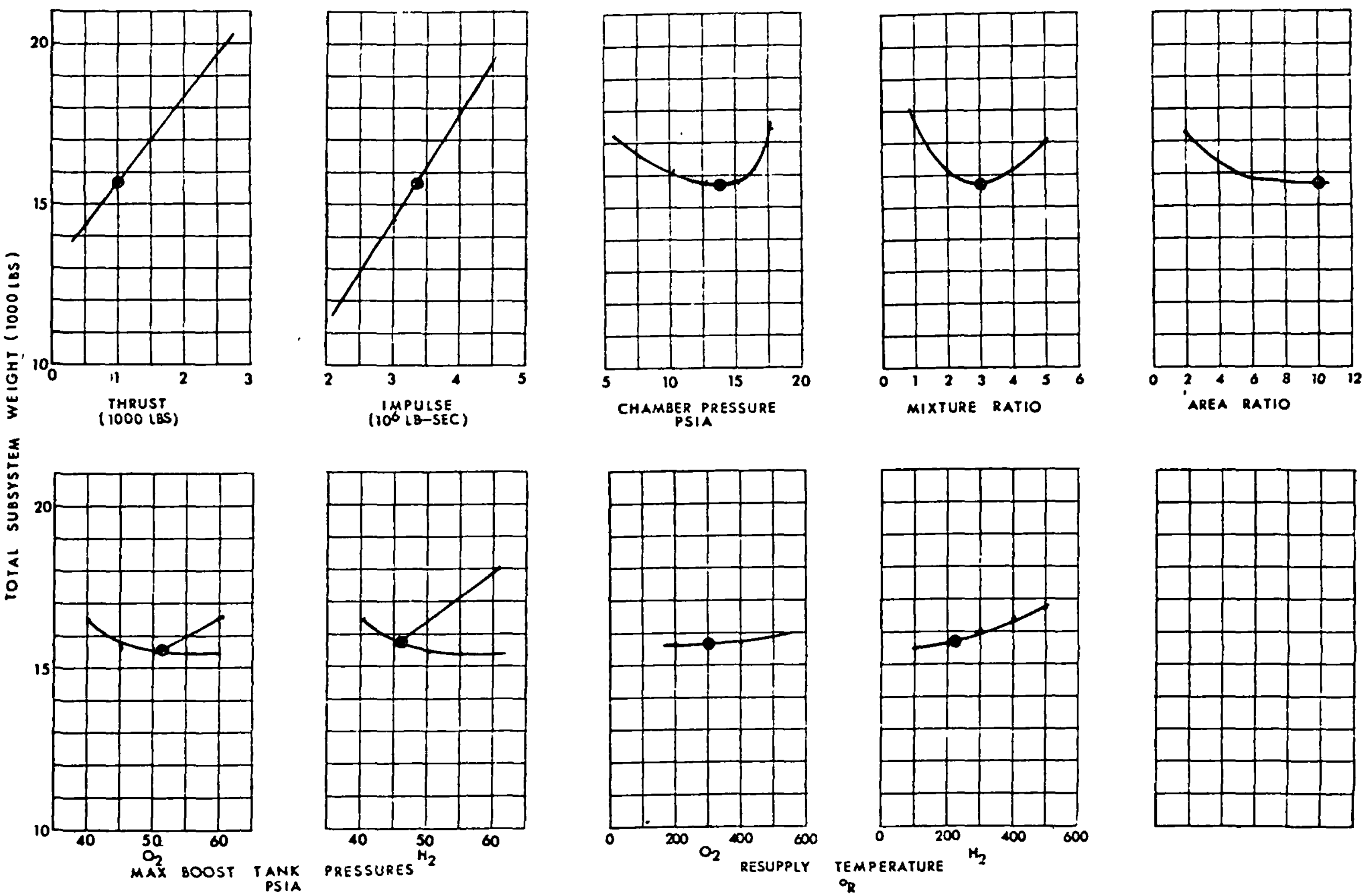
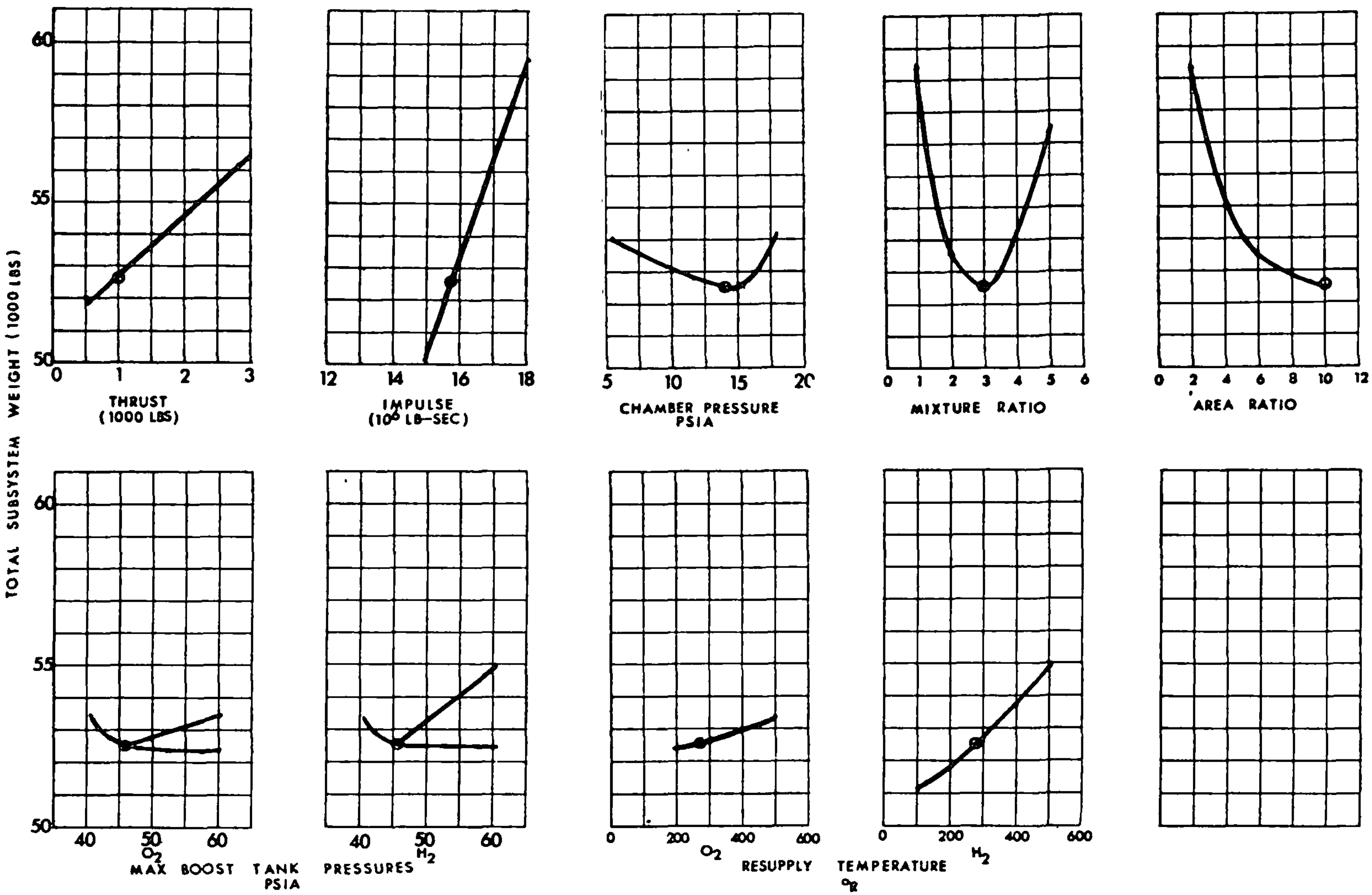


FIGURE F-48

LOW PRESSURE APS

ORBITER B (≤50 FPS)
SUPERCRITICAL– ACTIVE



LOW PRESSURE APS

ORBITER B (ALL MANEUVERS)
LIQUID— ACTIVE HEX

FIGURE F-49

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